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PARAFOIL POWERED FLIGHT PERFORMANCE

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Notre Dame University

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JOHN D. NICOLAIDES

UNIVERSITY OF NOTRE DAME

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PARAFOIL POWERED FLIGHT PERFORMANCE

JOHN D. NICOLAIDES

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FOREWORD

This report was prepared by the University of Notre Dame, Notre Dame, Indiana under U. S. Air Force Contract F33615-71-C-1093. This contract was initiated under Project 6065, Performance and Design of Deployable Aerodynamic Decelerators, Task 6065 01, Terminal Descent Parachutes for Tactical Air Drop and Military Vehicle Recovery. The work was administered under the direction of the Recovery and Crew Station Branch (AFFDL/FER) of the Air Force Flight Dynamics Laboratory at Wright-Patterson Air Force Base, Ohio. Mr. R. Speelman served as project engineer during the duration of the effort.

The author, of the University of Notre Dame Aerospace and Mechanical Engineering Department, was Dr. John D. Nicolaides, Professor. Contributing students of the University of Notre Dame Aerospace and Mechanical Engineering Department were, Michael Tragarz, Michael Higgins, Patrick Damiani, and Ed Tavares.

This report was released by the author in January 1972.

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Publication of this report does not constitute Air Force approval of the report's findings or conclusions. It is published only for the exchange and stimulation of ideas.

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ABSTRACT

The predicted flight performance of a powered Parafoil flight vehicle is calculated from solutions which are obtained from the Parafoil equations of motion. Flight vehicle total weights of 350, 400, 500, and 540 pounds are considered. Parafoil wing areas of 200 square feet and 400 square feet are considered. Wing loadings include .875, 1.0, 1.25, 1.35, 1.75, 2.0, and 2.7 pounds per square foot. Steady state flight trim angles of attack cover a range from -6^o to $+ 80^o$. The flight performance analyses include level flight, climbing flight, and descending flight. The computed flight parameters include the total velocity, the rate of climb (sink), the angle of climb (descent), and the horsepower required for the type of flight under consideration. The calculations suggest that powered Parafoil flight is possible. Actual piloted powered Parafoil flights demonstrate this possibility and confirm the feasibility. Various applications are suggested.

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LIST OF SYMBOLS

α	angle of attack (deg)
α_T	trim angle of attack (deg)
γ	angle that the flight path makes with the horizontal (deg)
η	dimensionless thrust factor
θ	thrust angle; angle that the thrust line makes with the horizontal (deg)
ρ	density of air (slugs/ft^3)
A	planform area of Parafoil airfoil (ft^2)
AR	aspect ratio
BHP	brake horsepower of engine
BHP_h	brake horsepower at altitude
BHP_o	brake horsepower at sea level
C_D	coefficient of drag
ΔC_D	additional drag coefficient due to size of cart
C_L	coefficient of lift
D	drag force (lbs)
ΔD	additional drag due to size of cart (lbs)
deg	degrees
fpm	feet per minute
fps	feet per second
ft	foot or feet
g	gravity constant (32.2 ft/sec^2)
HP	horsepower
HP_A	horsepower available

LIST OF SYMBOLS (continued)

HP_R	horsepower required for level flight at a given α_T
HP_X	excess horsepower for calculation of rate of climb ($=HP_A \cos\theta - HP_R$)
L	lift force (lbs)
lbs	pounds force
L/D	aerodynamic lift to drag ratio
$(L/D)_s$	glide ratio of total system in zero winds (system L/D)
$(L/D)_E$	glide ratio of total system inertial glide path in a specified wind (effective L/D)
m	mass
mph	miles per hour
ND 2.0 (400)	indicates a Parafoil with an aspect ratio of 2.0 and a planform area of 400 ft^2
P_h	pressure at altitude (lbs/ft^2)
P_o	pressure at sea level (lbs/ft^2)
R/C	rate of climb available under conditions being analyzed (fpm)
T	thrust (lbs)
Δt	time step (seconds)
T_A	thrust available (lbs)
T_h	temperature at altitude (degrees Rankine)
T_o	temperature at sea level (degrees Rankine)
T_R	thrust required to maintain level flight (lbs)
u	horizontal velocity; velocity in x direction (fps)

LIST OF SYMBOLS (concluded)

V	total velocity (fps, mph)
w	vertical velocity; velocity in z direction (fps)
W	weight (= mg) (lbs)
x	horizontal inertial axis
$\ddot{x}$	acceleration along x axis
z	vertical inertial axis
$\ddot{z}$	acceleration along z axis

SECTION I

INTRODUCTION

General

The predicted flight performance of various powered Parafoil flight vehicles is presented in this report which is prepared for the U. S. Air Force Flight Dynamics Laboratory under Contract No. F33615-71-C-1093.

Also, included are some preliminary results from the actual flights of various versions of a piloted powered Parafoil flight test vehicle called the "Irish Flyer".

In the sections which follow a brief background is given for the university, the Parafoil, the flight equations, and the performance results. Some powered Parafoil applications are suggested.

Early Aviation Interests

Before the advent of the airplane in 1903, the University of Notre Dame had already set forth the basic criteria for efficient aeronautical flight (L/D)[1], had carried out actual free flight tests of gliding models of birds, squirrels, and aircraft forms,[2] had developed the principles of soaring[3], had established the basic requirements for stable aircraft flight and control[4], and had constructed various aeronautical test equipment including the first prototype wind tunnel in the United States.[2,5] The interest of the university in aviation has continued unabated over the years.[6,7] The Department of Aeronautical Engineering was established in 1935,[7] the Department of Aero-Space Engineering was established in 1964,[8] and the Department of Aero-space and Mechanical Engineering was established in 1969.

Multi-Cell Kite

In December of 1964 the Multi-Cell Kite* was tested at the university. These tests included kite tests, wind tunnel smoke flow observations and aerodynamic measurements on a cut down unit. The unique ram air wing principle was established and applied to the design of the Parafoil by Professor Nicolaides, Figure 1.[9-19]

*Patent No. 3285546.

Parafoil

The Parafoil* is a flying wing with an airfoil section and a rectangular planform, Figure 1. It is made entirely of nylon cloth and, therefore, it differs from the conventional aviation wing in the very important feature that it is completely nonrigid. Thus, it can be packed and deployed like a conventional parachute. The Parafoil obtains its rigidized flight configuration from the ram air pressure entering the large openings in the leading edge. It is composed of individual air cells connected by porous cloth ribs to allow pressure equalization throughout the interior. The exterior is made of a low porosity nylon fabric. Therefore, the air in each cell and in the Parafoil as a whole is essentially stagnant and ram air pressurized. The pennants along the bottom surface serve to distribute the aerodynamic and payload forces uniformly along the bottom surface. They also reduce the aerodynamic losses at the tips of the unit.[14]

The Parafoil, therefore, is really an aircraft or glider which can be packed in a small unit and deployed when needed. In flight, it performs like the conventional wing of aviation and, thus, it may be considered for many applications not heretofore possible. Some of the applications for which Parafoils have been constructed or proposed are:

- Pilot Recovery[20]
- Manned Jump[13,15,17,22]
- Guided and Controlled Delivery Systems[22,23]
- Underwater Delivery Systems[24]
- Munitions Delivery Systems[12,25]
- Space Capsule Recovery[26]
- Kite Flight[15]
- Decoy and Countermeasure Systems
- Homing Destruction Systems
- and others

Powered Parafoil

Early in the flight test program the Parafoil was attached to a cart and towed aloft to altitudes of 500 feet and 1,000 feet, Figure 2. When the tow line was released, the cart with the Parafoil would glide to earth, Figure 3. By measuring the glide angle and the gliding velocity, the lift-to-drag ratio and the aerodynamic coefficients of the Parafoil were determined in much the same way that Professor Zahm had done almost a century earlier on the campus. The gentleness and stability of these cart flights lead to the introduction of a pilot, Figure 4, and an engine, Figure 5.[21]

*The Parafoil is a design and development by Dr. J. D. Nicolaides (Patent Pending No. 105,836).

By 1970 it was clear that additional attention should be given to powered Parafoil flight both because of the advances in Parafoil technology and because of the emerging importance of Parafoil applications to powered pilot recovery, stand-off weapons delivery, and other areas. Accordingly, the University requested the U. S. Air Force Flight Dynamics Laboratory to make available some of its flight vehicles for powered Parafoil flight tests. Such an arrangement could not be made. However, the Flight Dynamics Laboratory continued to be interested in the concept of powered Parafoil flight and under Contract F33615-71-C-1093 provided support for performance calculations. This report provides the results of these calculations and, also, provides some experimental validations of the calculations.

SECTION II

THEORY OF PARAFOIL POWERED FLIGHT

The flight of a Parafoil differs from the flight of an airplane in that it can fly over a wide range of trim angles of attack from -6° to $+ 80^{\circ}$. Also, in an aircraft the wing is rigidly attached to the fuselage and, thus, it pitches, yaws, and rolls with the aircraft. In the case of the powered Parafoil vehicle, the vehicle maintains its angle of pitch independent of the pitch of the Parafoil. Therefore, in considering the flight performance of a powered Parafoil, it is necessary to formulate suitable equations of motion (1) in the case of small angles of trim and climb, and also (2) in the case of large angles of trim, pitch, and climb. Further, the thrust line is fixed to the vehicle and not to the Parafoil and, thus, its line of action can be at a small or large angle to the horizon as may be desired for obtaining optimum flight performance. In the two sections which follow the equations of motion for steady state Parafoil flight are formulated for small angle flight and for large angle flight.

Small Angle Flight Theory

The general equations for Parafoil flight are given by, Figure 6.

$$T\cos\theta + L\sin\gamma - D\cos\gamma = m\ddot{x} \qquad (1)$$

$$-T\sin\theta - L\cos\gamma - D\sin\gamma + mg = m\ddot{z} \qquad (2)$$

For level steady state flight, Equations (1) and (2) reduce to

$$T\cos\theta - D = 0 \qquad (3)$$

$$-T\sin\theta - L + mg = 0 \qquad (4)$$

The total velocity of the Irish Flyer in level flight is obtained from Equations (3) and (4) as:

$$V = \sqrt{\frac{2W}{\rho A}\left(\frac{1}{C_L + C_D\tan\theta}\right)} \qquad (5)$$

This value of velocity substituted into Equation (3) yields the thrust required for level flight:

$$T_R = \frac{W\,C_D}{(C_L\cos\theta + C_D\sin\theta)} \qquad (6)$$

The horsepower required for steady state level flight[27] (using Equation 3),* and the thrust available are given as

$$HP_R \equiv \frac{DV}{550} = \frac{T_R V\cos\theta}{550} \quad (7a) \qquad HP_A = \frac{T_A V}{550} \qquad (7b)$$

For small flight angles the rate of climb is given by,

$$R/C = \frac{HP_A \cos\theta - HP_R}{W} \; 33{,}000 \quad \text{(feet/minute)} \qquad (8)$$

$$\gamma \text{ small}$$

By utilizing Equations (5) through (8) together with wind tunnel values for the aerodynamic coefficients, $C_L(\alpha)$ and $C_D(\alpha)$, the flight velocity and horsepower required for Parafoil steady state level flight may be obtained.

Large Angle Flight Theory

The Parafoil is able to achieve trim angles of attack, α_T, from -6^o to $+80^o$ and can achieve flight angles, γ, from 90^o to above -40^o. Thus, it is essential to consider the full equations. For steady state flight, Equations (1) and (2) may be written as

$$T\cos\theta + C_L \frac{1}{2}\rho V^2 A\sin\gamma - C_D \frac{1}{2}\rho V^2 A\cos\gamma = 0 \qquad (9)$$

$$-T\sin\theta - C_L \frac{1}{2}\rho V^2 A\cos\gamma - C_D \frac{1}{2}\rho V^2 A\sin\gamma + W = 0 \qquad (10)$$

Solving Equations (9) and (10) for the total velocity yields:

$$V^2 = \frac{T\cos\theta}{(C_D\cos\gamma - C_L\sin\gamma)1/2\,\rho A} \qquad (11)$$

---

*For unpowered gliding flight we may write:

$$V \approx u \approx w \;(L/D) \qquad (\text{for } L/D > \sim 3) \qquad (5a)$$

Substituting this equation into Equation (7a) and noting that $L \approx W$ we obtain:

$$HP_R \approx \frac{Ww}{550} \qquad \theta \text{ small} \qquad (7c)$$

This equation is helpful in utilizing gliding flight test results in order to obtain an estimate of the horsepower required for level flight since the rate of sink, w, is measured relatively easily.

$$V^2 = \frac{W - T\sin\theta}{\frac{1}{2}\rho A\,(C_L\cos\gamma + C_D\sin\gamma)} \tag{12}$$

Equating Equations (11) and (12) yields:

$$\tan\gamma = \frac{\left(1 - \frac{T\sin\theta}{W}\right) - \frac{L}{D}\left(\frac{T\cos\theta}{W}\right)}{L/D\left(1 - \frac{T\sin\theta}{W}\right) + \left(\frac{T\cos\theta}{W}\right)} \tag{13}$$

By defining the mathematical quantity,

$$\eta = \frac{T}{W/(L/D)} \tag{14}$$

we may simplify Equation (13) as*

$$\gamma = \tan^{-1}\frac{\left(1 - \frac{\eta\sin\theta}{L/D}\right) - \eta\cos\theta}{L/D\left(1 - \frac{\eta\sin\theta}{L/D}\right) + \frac{\eta\cos\theta}{L/D}} \tag{15}$$

and Equation (12) may be written as,

$$V = \sqrt{\frac{W - \frac{\eta W}{L/D}\sin\theta}{\frac{1}{2}\rho A\,(C_L\cos\gamma + C_D\sin\gamma)}} \tag{16}$$

where

$$u = V\cos\gamma \tag{17}$$

$$w = V\sin\gamma = -\frac{R/C}{60} \tag{18}$$

Thus, the flight path angle, γ, of the Irish Flyer may be obtained from Equation (15) by inputting the numerical value of the thrust angle (θ), the lift-to-drag ratio (L/D) for a fixed flight trim angle of attack (α), and the thrust factor η. The total velocity of the Irish Flyer may then be obtained from Equation (16) by inputting γ as obtained from Equation (15) and $C_L(\alpha)$ and $C_D(\alpha)$. The horsepower required may now be obtained by utilizing Equation (7).

Thus, we are able to obtain the flight performance of the powered Parafoil from solutions of the large angle equations of motion.

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*It may be noted in Equation (15) that when $\theta = 0$ and $\eta = 1$ level flight is achieved.

SECTION III

FLIGHT PERFORMANCE CALCULATIONS

The first performance calculations were carried out on a 400 pound powered Parafoil vehicle in level flight utilizing a 200 sq. ft. Parafoil and a 400 sq. ft. Parafoil.

The basic aerodynamic coefficient data, $C_L(\alpha)$, $C_D(\alpha)$, used is given in Ref.14, and is presented in Figure 7. This data includes the drag of the isolated Parafoil, the drag of the suspension lines (C_{D_L} = 0.016, based on a total line area of 5.5 ft^2 and a drag coefficient of .6), and the drag of a small payload (C_{D_m} = 0.010, based on an area of 2.5 ft^2 and a drag coefficient of .8), all associated with the 200 sq. ft. Parafoil. For the powered Parafoil vehicle computations using the 200 sq. ft. Parafoil, the data in Figure 7 was modified by adding an additional vehicle drag of $\Delta C_D = +0.076$ (based on an additional vehicle area of 19 sq. ft. and a drag coefficient of .8). Therefore, the aerodynamic data employed includes the effects of the Parafoil, the lines, and a vehicle having an area of 21.5 ft^2.

In the case of the 400 sq. ft. Parafoil the data of Figure 7 was again used and the added vehicle drag was reduced by 1/2 thus yielding a ΔC_D = .038.

In carrying out various computer studies both values of incremental drag were actually utilized for both sizes of Parafoils and for various total system weights, so as to provide a more general parametric study.

Level Flight

Sea Level Flight

The level flight performance calculations are carried out using the small angle flight theory equations. Also, included is an estimate of the potential rate of climb, Eq.(8), based on the horsepower available in excess of that required for level flight.

For a Parafoil area of 400 square feet, curves for $V(\alpha)$, $HP_R(\alpha)$, $R/C(\alpha)$, $HP_R(V)$, and $R/C(V)$ using 24 horsepower are given in Figures 8 and 9. The seme performance factors are given in Figures 9 and 10, for a Parafoil area of 200 square feet using a $\Delta C_D = +.076$. Thus, the calculations for three wing loadings, 1.0, 1.25, and 2.0, for two incremental drags, ΔC_D=.038 and .076 are given in Tables I-V.

Altitude Flight

Performance calculations for level flight were also carried out for altitudes of 5000 and 10,000 feet,* (Tables VI and VII). The service ceiling** of the Parafoil is approximately 17,000 feet using $BHP_o = 24$.

The reduction of engine performance with altitude was taken into account. The equation used to determine the brake horsepower available at altitude is:[28]

$$BHP_h = BHP_o \left(\frac{P_h}{P_o}\right)^{1.15} \left(\frac{T_h}{T_o}\right)^{-0.5}$$

Figure 11 is a plot of maximum rate of climb versus altitude for the powered Parafoil flight vehicle with a wing loading of one, $(W/A = \frac{400}{400})$.

Irish Flyer***

A prototype powered flight vehicle was constructed as a test platform for investigating the various design variables such as engine size and weight, thrust angle, vehicle weight, L/D, etc. The total weight of this vehicle including the pilot is 540 pounds. Accordingly, flight performance calculations were carried out for this experimental flight vehicle weight using both a 400 square foot Parafoil ($W/A = 1.35$) and a 200 square foot Parafoil ($W/A=2.7$). A 350 pound vehicle was also considered. Calculated level flight results for $V(\alpha)$ and $HP_R(\alpha)$ are given in Figure 12. $HP_R(V)$ is given in Figure 13. Figures 14 and 9a present $HP_R(V)$ for various Irish Flyer weights and wing loadings, $(\frac{350}{400} = .875, \frac{350}{200} = 1.75, \frac{400}{400} = 1, \text{ and } \frac{400}{200} = 2)$.

Thrust Angle

Since the pitch angle of the cart and the trim angle of the Parafoil are independent and since the engine and propeller line of thrust may be fixed at different angles to the horizon, special flight performance calculations were carried out for thrust angles of $-20^o, -10^o, 0^o, 10^o, 20^o, 30^o$, and 40^o for a wing loading of 1.0 $(W/A = \frac{400}{400})$, and an additional drag of $\Delta C_D = .038$, Tables VIII - XIII.

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*These calculations assume a wing loading of one on the 400 square foot Parafoil with a ΔC_D of 0.038.

**Service Ceiling - ceiling at which the rate of climb is 100 (fpm) for a specified HP_A.

***All rights to powered Parafoil applications and to Irish Flyer concept are held by John D. Nicolaides.

Climbing and Descending Flight

The performance calculations, Equations (5), (6), (7), and (8) of the previous section were all for small angle and level flight. These calculations showed, however, that large rates of climb were possible; so large, in fact, that the small angle assumptions were no longer valid. Accordingly, exact computations, using the large angle equations are now carried out in this section for climbing and descending flight; which also include the case of level flight, $\gamma = 0$.

The flight performance of the 400 pound flight vehicle using the 400 square foot (W/A=1) Parafoil is calculated for a fixed angle of trim ($\alpha_T = 11^o$), and the additional drag of $\Delta C_D = .076$ (L/D=2.95). The calculations include thrust angles of $0^o, 8^o, 16^o$ and 24^o. The flight parameters HP (R/C), HP(γ), and HP(V) are given in Figure 15.* Table XIV provides flight parameters for various values of η at $\theta = 0$.

Flight performance calculations were also carried out for a 540 pound prototype flight vehicle again using the 400 and 200 ft^2 Parafoils. The results for HP (R/C), HP(γ) and HP(V) are given in Figures 16* and 17*. Also see Tables XV and XVI.

Constant Horsepower Performance

The performance calculations of the previous section utilized Eq. (7), (15) and (16) which yield the horsepower required for various flight modes. It is possible to input the horsepower available as a constant and then to solve for the various flight performance parameters by iteration of the flight equations. Representative results for V (α), $\gamma(\alpha)$, and R/C (α) are plotted in Figure 18 for a Parafoil area of 400 square feet, for a flight vehicle weight of 540 pounds ($\Delta C_D = .076$) and for horsepowers of 20, 30, and 40. Summary data is given in Figure 18d and Table XVII.

Performance calculations are also carried out for the 540 pound flight vehicle using a 200 square foot Parafoil, Figure 19. A summary curve is given in Figure 19d.

The effects of thrust angle on the 400 ft^2 Parafoil with a 540 pound payload are shown in Figure 20 for a constant horsepower of 20.

*Figures 15, 16 and 17 are approximations and should not be used for detail design analysis.

SECTION IV

DISCUSSION OF PERFORMANCE PREDICTIONS

Level Flight

The effects of flight vehicle weight, Parafoil wing area, trim angle of attack, thrust line of action, and additional vehicle drag are readily seen in the figures and tables. For example, for the 540 pound vehicle using the 400 square foot ($W/A = 1.35$) Parafoil, a trim angle of attack near 10° provides minimum horsepower required. See Figures 12, 13, 16 and 18. The horsepower required for level flight is approximately 12 HP and the flight velocity is 37 feet per second or 25 miles per hour. The flight velocity may be increased by reducing the trim angle of attack. At a trim angle of 0° the level flight velocity is approximately 54 feet a second or 37 miles per hour and the horsepower required is 30. It is noted, Figure 16a, that elevation of the thrust line of action reduces the horsepower required to 10 HP for level flight at $\alpha = 11^{\circ}$.

Ascending Flight

Again using the 540 pound vehicle with the 400 square foot wing area as an example, we note from Figure 18a that for a trim angle of attack near 10° the rate of climb is 450 feet per minute and the climb angle (γ) is 11° using 20 horsepower. Using 30 horsepower we obtain from Figure 18b a rate of climb of 1050 feet per minute and an angle of climb of 21°. A substantial reduction in required horsepower may be obtained by elevating the line of thrust, particularly at the higher rates of climb, Figure 16 and 20.

These values for powered Parafoil flight performance are achieved because of the small weight of the Irish Flyer. This small weight is achieved due to (1) the very light wing (the 400 ft^2 Parafoil weight is only 15 pounds) and (2) the light fuselage which does not have to resist any aerodynamic bending moments as does an aircraft which has rigid wing and rigid elevators.

SECTION V

FLIGHT PERFORMANCE TESTS*

The flight performance tests of the powered Parafoil vehicle were composed of two phases.

Phase I is composed of gliding flight tests which are achieved by towing the vehicle to an altitude from 500 feet to 1000 feet and then releasing it so that it can glide freely back to earth. Measurements are taken of the steady state gliding flight. Both unmanned and manned flights were carried out.

Phase II is composed of powered flight tests and is carried out in a manner similar to Phase I except that the engine is running.

Flight Test Vehicle

The flight test vehicle used in the test program was named the Irish Flyer II** and is shown in Figure 21. It was designed so as to provide a safe and versatile flight platform for investigating the various vehicle design parameters such as engine types, engine location, engine angle, Parafoil size, Parafoil attachment, Parafoil controls, center of gravity location, wheel base, etc. The weight of the Irish Flyer with pilot is 540 pounds. The Parafoil ND 2.0 (400) was used which has an aspect ratio of 2.0 and an area of 400 square feet. The horsepower of the rebuilt Volkswagen engine is supposed to be 28 HP; however, the actual horsepower available by static test is estimated to be only 12 HP due to low engine RPM, constant spark advance, and low propeller efficiency.

Irish Flyer II Physical Characteristics

Parafoil (ND 2.0 (400))	400'
Vehicle overall length	10'10"
Vehicle height (without canopy)	4' 7-1/2"
Vehicle height (with canopy)	33'3"
Vehicle width (without canopy)	6'1"
Vehicle width (with canopy)	28'4"
Propeller Diameter	4'6"
Wheel Base	5'9-1/2"
Width of Parafoil attachment points	5'10"
Weight engine	131 pounds
Empty weight	323 pounds
Gross weight	540 pounds
Useful horsepower (estimated)	12 ± 3

*Dr. John D. Nicolaides acting completely on his own authority and responsibility undertook the design and construction of the flight vehicle and carried out the associated flight test program.

**The FAA/SAC of 20 July 1971 assigns N-3029 to "Nicolaides-Parafoil Flyer."

Control System

The Parafoil is attached to the vehicle on the outside ends of the horizontal bar on the top of the vehicle. Originally, the control system of the cart was attached to the rear control lines of the Parafoil giving a limited capability to turn and the capability for a full flare. The wires of the control system are strung so as to give a two to one deflection for turning with a full deflection of approximately eighteen inches and a three to one deflection for flaring with a full flare potential of five and one half feet deflection. The flare is actuated by pushing a foot lever forward with both feet to the extension desired. It is estimated that the force required to throw this lever is approximately fifty pounds

The original control system designed for the vehicle allowed for turning control by pulling down either side of the rear control lines with a two to one deflection by the turning of handle-bar type device by the pilot. The maximum deflection afforded by this system was eighteen inches and the initial flight tests showed that this deflection on the four hundred square foot canopy was not sufficient to allow proper turn control. The time required to make a 90^{o} turn was approximately 20 seconds. To overcome this, a separate control system was incorporated which made use of the magic flare control* of the Parafoil canopy. It had previously been determined that the use of the magic flare allowed turn control with much smaller deflection. A magic flare type of control was added to the previous type of control. This magic flare control was designed so as to give a two to one ratio of deflection through the use of a sliding control lever. With this new turn control system, a ten inch deflection by the pilot produces a 20" deflection at the canopy which provides for a more than adequate control response. With the use of the magic flare control system, the time required to make a 90^{o} turn was reduced from 20 to 5 seconds.

Flight Test Results

Gliding Flight

Various instruments were utilized in the gliding flight tests. Some instruments were mounted on the flight vehicle which provided the rate of climb, rate of sink and total velocity. The instrument readings were taken by the pilot during the flight and recorded immediately afterwards. Also, a movie camera was strapped to the rear of the flight vehicle which photographed the instrument readings, the control deflections of the pilot, the response of the vehicle, and provided a dramatic view of the in-flight stability and safety of the vehicle. The primary data used was obtained from a movie camera located on the ground down range of the launch and so situated that the flight path was approximately perpendicular to the line of sight of the camera during analysis. A vertical reference marker was placed in the

*The magic flare control system consists of a line from the pilot to the third flare back in the second row of flares inboard from each side.

field of view. Smoke grenades were attached to the flight vehicle and ignited by the pilot during the ascending portion of the flight. By measuring the angle of the smoke trail, the lift to drag ratio of the gliding system was determined.

The film from the ground camera was measured and yielded the flight path. The measurement of the smoke angle gives the system's L/D. The measurement of the flight path gives the effective L/D. From these two a check on the wind velocity can be made and compared to the wind velocity readings made prior to the flight. Using the smoke trail as the direction of the velocity vector of the flight vehicle and the orientation of the Parafoil, the trim angle of attack of the Parafoil is measured.

By measuring the distance between two reference points a known distance apart on a frame, a length dimension factor was obtained. The true distance that the flight vehicle descends between two frames can then be determined using this length dimension factor and a common reference point. Knowing the frame rate of the camera and counting the number of frames between the two frames on which the descent is measured, the time of descent can then be obtained and the rate of sink calculated. Multiplying the rate of sink by the lift to drag ratio from the smoke gives the no wind horizontal velocity. Then, knowing the vertical and horizontal velocities, the total velocity can be calculated.

Figure 22 is a picture of one of the data frames on the ground camera data film. Figure 23 shows how the measurements of the first frame of data from the first flight were taken from the ground camera film. It is known that the distance between the near attachment point and the front side flare tip is 27 feet and by measuring this distance on the data film, the length dimension factor is obtained. Superimposed on this figure is the data from the other frames in the first flight. The line formed by these point locations shows the actual flight path of the flight vehicle.

A list of the data taken from the ground film on the first flight is given below. The rates of sink as shown were calculated over a time step of six data frames. Each data frame was taken on every fifth film frame, so the time step for each rate of sink was over thirty frames of film. The speed of the film was 24 frames per second, so the actual time of each time step in the calculation of the rate of sink was 1.25 second.

Data Frame No.	Smoke Angle (deg)	Smoke L/D	Angle of Attack (deg)	h* (in.)	l** (in.)	Attachment Point Height (ft.)	Rate of Sink (fps)
1	11.3	5.005	8.0	2.41	1.56	41.71	
2	13.2	4.264	9.6	2.30	1.63	38.09	
3	12.0	4.705	7.2	2.18	1.62	36.33	
4	10.3	5.503	5.0	2.08	1.67	33.62	12.14
5	11.3	5.005	5.5	1.96	1.62	32.66	10.02
6	10.9	5.193	4.8	1.85	1.65	30.27	8.77
7	13.1	4.297	8.3	1.70	1.73	26.53	8.09
8	12.0	4.705	7.8	1.60	1.69	25.56	9.10
9	15.0	3.732	11.0	1.55	1.65	25.36	8.36
10	11.0	5.145	4.0	1.48	1.70	23.50	
11	10.7	5.292	3.3	1.34	1.70	21.28	
12	13.1	4.297	4.7	1.27	1.73	19.82	

It is seen from Figure 23, that the plot of the flight path positions is a straight line with the actual flight path angle of 16.8 degrees. This yields a effective flight path L/D without wind correction of 3.312. The average system L/D calculated from the smoke angle is 4.761 which is the L/D with wind correction. The average rate of sink above is 9.41 feet per second. The vehicle weight was 409 pounds.

Using the system L/D from the smoke, yields a horizontal component velocity of 44.80 feet per second. Calculating the total velocity corrected for the wind, gives a value of 45.78 feet per second. Using the effective L/D of the actual flight path angle to calculate horizontal and total velocities yields values of 31.16 and 32.55 feet per second respectively. The difference in the horizontal components of velocity between 44.80 and 31.16 of 13.64 feet per second is the calculated wind velocity. In other words, according to the calculations, the flight vehicle was descending into a wind of 13.64 feet per second.

There were five flight test data runs performed on the flight vehicle with varying amounts of simulated engine weight ranging from forty to one hundred and twenty pounds. A number of flight tests had been performed previously without the simulated engine weight in order to evaluate the control response of the flight vehicle and its structural strength. In the preliminary flight tests it had been determined that the nose wheel as shown in Figures 1 and 21 was too small and this was replaced by a wheel of larger diameter and tread width to support the weight. Structurally, the flight vehicle checked

*h - height of attachment point above a reference point as measured on film.

**l - distance from front outside flare tip to attachment point as measured on film.

out to be quite adequate and after the addition of the magic flare control system, the data tests with the additional simulated engine weight were performed.

Of the five data flights only the first and fifth flights provide reducible data for analysis. These data are:

Gliding Flight	1	5
Total Weight (lbs)	409	492
Measured Wind (mph)	5-10	5-10
Smoke Angle (deg)	11.9^{o}	12.5^{o}
System L/D (Smoke)	4.76	4.51
Flight Path Angle (deg)	16.8^{o}	15.0^{o}
Effective L/D (Path)	3.31	3.73
Calculated Wind (fps)	13.6	6.75
Flight Velocity (fps)	32.5	33.5
Flight Velocity (No Wind) fps	45.8	40.0
Rate of Sink (fps)	9.4	8.7
Horsepower Required Eq. (7a)	7.0	7.7
Angle of Attack	6.6^{o}	11.8^{o}

The most important parameter determined by the data flights was the rate of sink. Using the rate of sink of the first flight and Equation (7a), the horsepower required estimate is 7.0 which is in agreement with the predicted value in Figure 9a.

The flight parameters of the first data flight which do not compare well with the theoretical calculations are the system L/D and the flight velocity with wind correction. The free flight tests showed an L/D value of 4.76 from the smoke. This is compared to the maximum theoretical value of L/D of 3.66 using $\Delta C_D = .038$.

Using Equation (7a) to estimate the horsepower required from the rate of sink determined by the ground film data on the fifth flight, a value of 7.75 horsepower is obtained. Checking the theoretical calculations made for a total weight of 500 pounds, as compared to the actual weight of 492 pounds, it is found that the minimum value of horsepower required determined by the theoretical calculations is 8.622, or approximately one more than that estimated from the flight data. This small discrepancy can easily be explained by a slightly low value of rate of sink determined from the flight data coupled with the fact that the actual flight weight was eight pounds lighter than for the theoretical calculations.

The system L/D's measured from the flight data were higher than used in the theoretical calculations. This improvement in system L/D could be due to any combination of three factors. The first and most obvious factor is a possible error in data reduction. A slight error in measurement of the

smoke angle of approximately one degree could account for the difference. Another possibility is the existence of thermals and gusts over the field. Finally, there is the possibility that the wind tunnel data used to make the theoretical calculations could have been conservative.

Powered Flight

On 24 August 1971 five powered Parafoil flights were carried out at the Goshen Airport, Indiana, Figure 24.

First Flight

The powered flights were carried out in the same manner as the gliding flights except that the engine is idling. On the first flight the Irish Flyer was towed to an altitude of approximately 600 feet. A steady state tow continued for approximately 1/3 mile. No problems were countered. The pilot then applied full throttle and slack appeared in the tow line. The Irish Flyer was observed to be flying with no yaw or pitch; thus, an "O.K. to release tow" radio message was sent to the pilot who then released the tow line and flew to the end of the runway, (1/4mile)where he landed softly with a ground roll of approximately 10 feet. During his flight he was estimated to descend slowly (2-6 ft/sec); part of the flight he was able to fly level. He was able to turn the Irish Flyer approximately 45° to the left in correcting a slight cross wind. After landing he immediately cut the engine and flared the Parafoil so that it fell to the ground behind the vehicle.

Second Flight

The second flight was similar to the first except that after reaching an altitude of 600 feet the pilot immediately released the tow line and flew approximately 3/8 mile in slowly descending flight (2-6 ft/sec). Some icing of the carburetor occurred which is believed to have reduced the useful horsepower. Landing was soft (2 ft/sec) with little landing roll, (5-10').

Third Flight

The third flight was similar to the second except with less icing due to increased temperature and thus more power was available. Right and left turns of 45° were executed with no difficulties. The Irish Flyer again flew with complete stability about all axes. Near level flight was again achieved. The distance of the flight was about 1/2 mile.

The landing was carried out with full throttle. The Irish Flyer touched down and then took off again flying approximately 25 feet before executing a normal landing with reduced power, with engine throttled back on touch down, and with Parafoil flare.

Fourth Flight

Prior to the fourth flight the engine was ground tested and the magneto was adjusted so as to provide better RPM. The wind had changed from North to West and thus a new runway was used. After release the pilot reported a climb from 600 feet to 1000 feet. From the ground the Irish Flyer was observed to climb and fly level for a distance of 3/4 mile. Again small turns were easily accomplished. At the end of flight the Irish Flyer flew level at about a 50 foot altitude for 10 to 15 seconds and about 500 feet. It was possible for the tow car to drive directly underneath and observe rigging, turn, control, etc. The landing was normal.

Fifth Flight

The last flight was similar to the fourth. The Irish Flyer exhibited complete stability and the pilot reported no need at all for rudder control, even in turns. Landing was normal. Distance from release to touchdown was approximately 1/2 mile.

Discussion of Results*

The flight performance of the powered Parafoil vehicle was nominal and as predicted. The horsepower available allowed straight flights of 1/2 to 1 miles distance. Flight stability and control was demonstrated as observed in the documentary moving picture films and as seen by the various observers. Landings were extremely soft (1-2 fps) and short.

*See Appendix B for results on Irish Flyer III.

SECTION VI

FUTURE APPLICATIONS

The flight tests have validated the performance predictions and have demonstrated the feasibility of stable and controlled powered Parafoil flight. These flight demonstrations now open an entirely new field of potential applications. Some of these are:

- Pilot Recovery and Return to Base
- Stand-Off Delivery of Troops (both individual and mass)
- Stand-Off Delivery of Cargo and Supplies (manned and guided)
- Stand-Off Delivery of Bombs (guided or homed, Remotely Piloted Vehicle)
- Rescue of Troops and Equipment
- Flying Jeep
- Air Drop Systems (aircraft or helicopters)
- Terminal Powered Guidance of Shells, Rockets, and Re-Entry Bodies.

SECTION VII

CONCLUSIONS

The flight performance of a powered Parafoil vehicle is predictable from the aerodynamic data obtained on the Parafoil canopy and lines. Actual flight of a powered Parafoil vehicle is obtainable as evidenced by preliminary powered flight tests. In these tests both level and climbing flight were demonstrated, and the flight performance appeared to match the predicted performance although more data is needed to confirm the relationship.

The analysis suggests that a change in pitch attitude of the powered Parafoil vehicle can increase its rate of climb and lower its level flight horsepower requirements.

Preliminary tests indicate that a more comprehensive program of testing is feasible.

Figure 1. Irish Flyer

Figure 2. Tow Ascending Flights

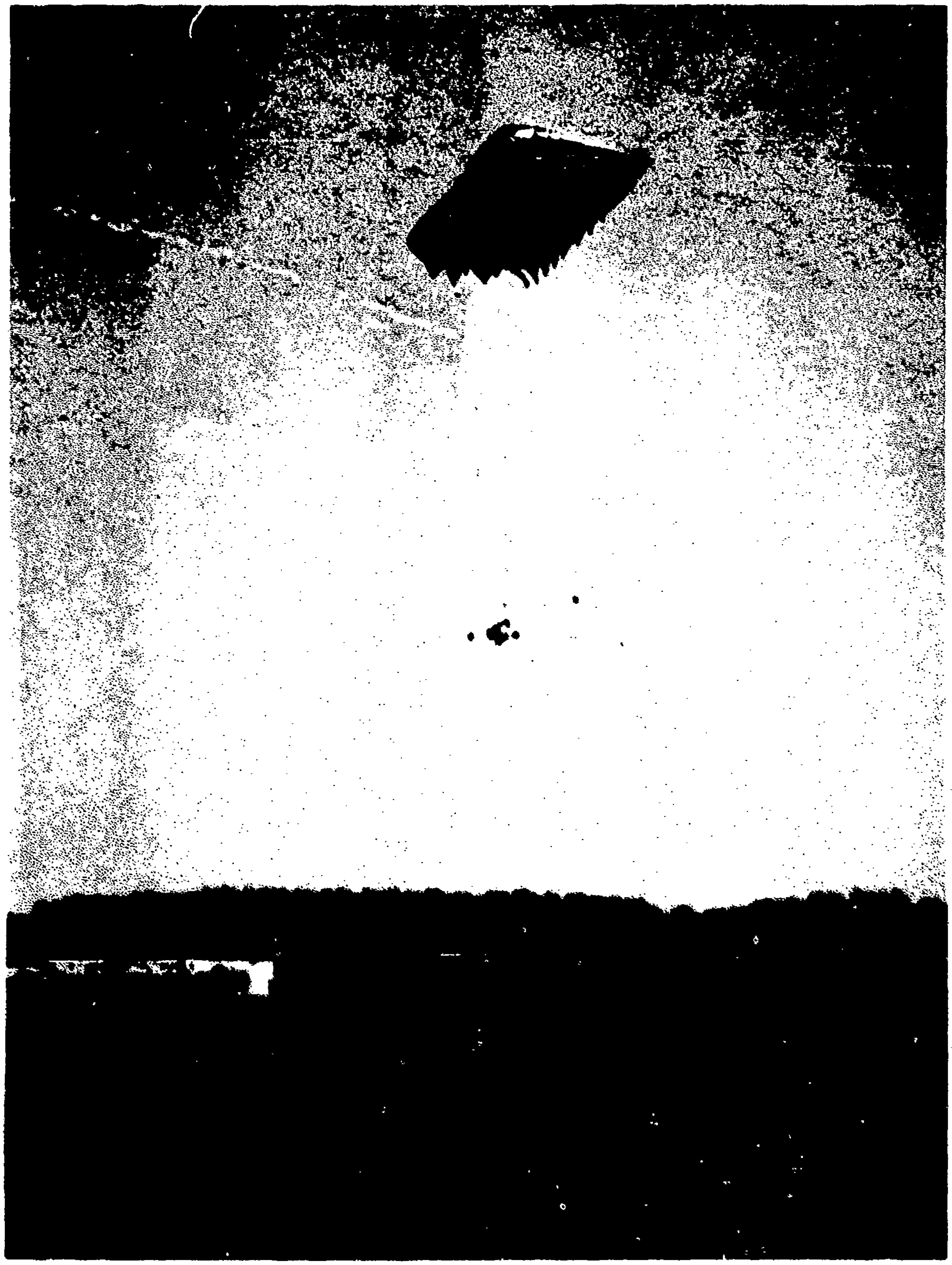
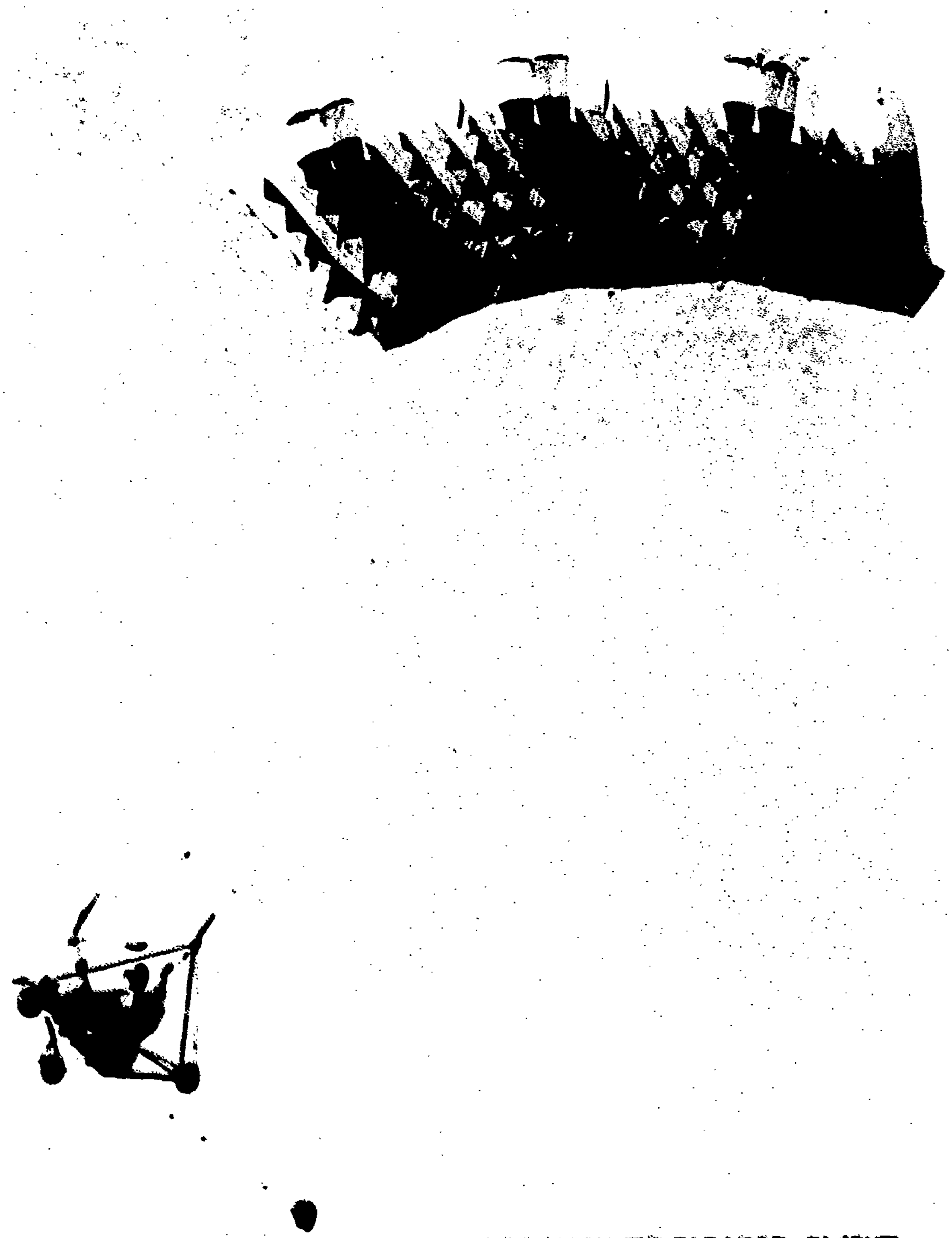


Figure 3. Parafoil Glider with 864 ft^2 Area



FIRST MANNED PARAFOIL FLIGHT
Figure 4

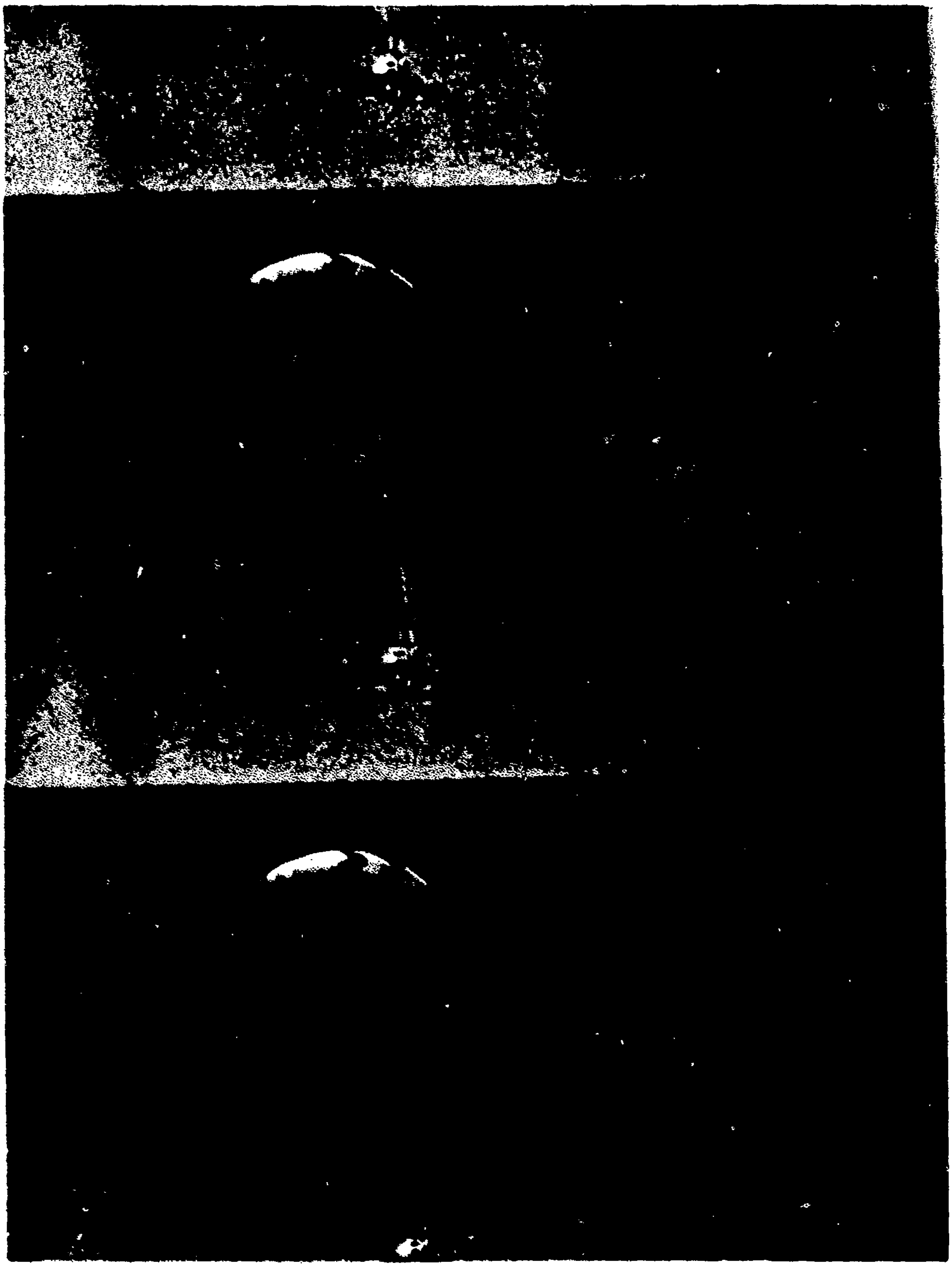
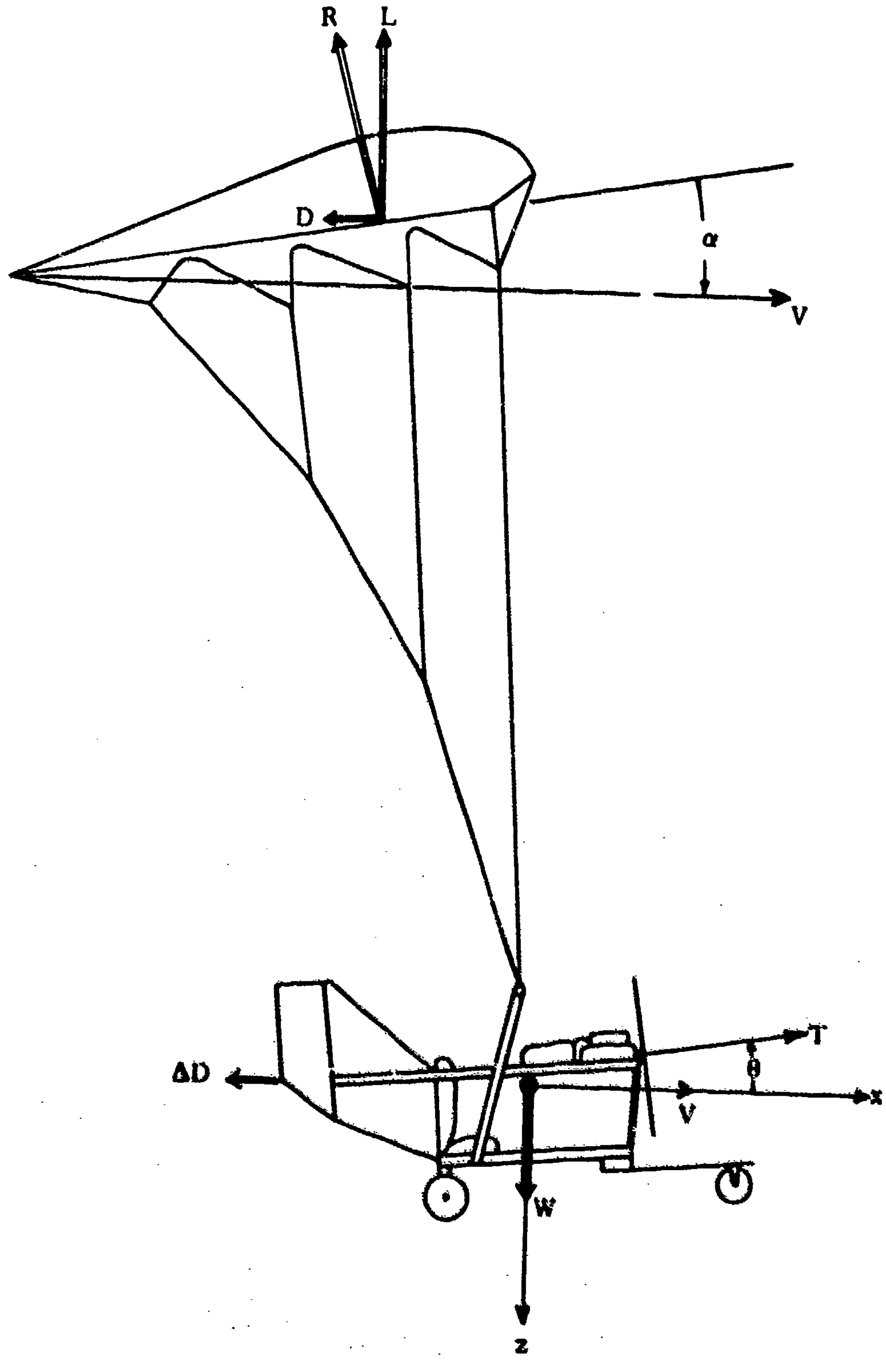
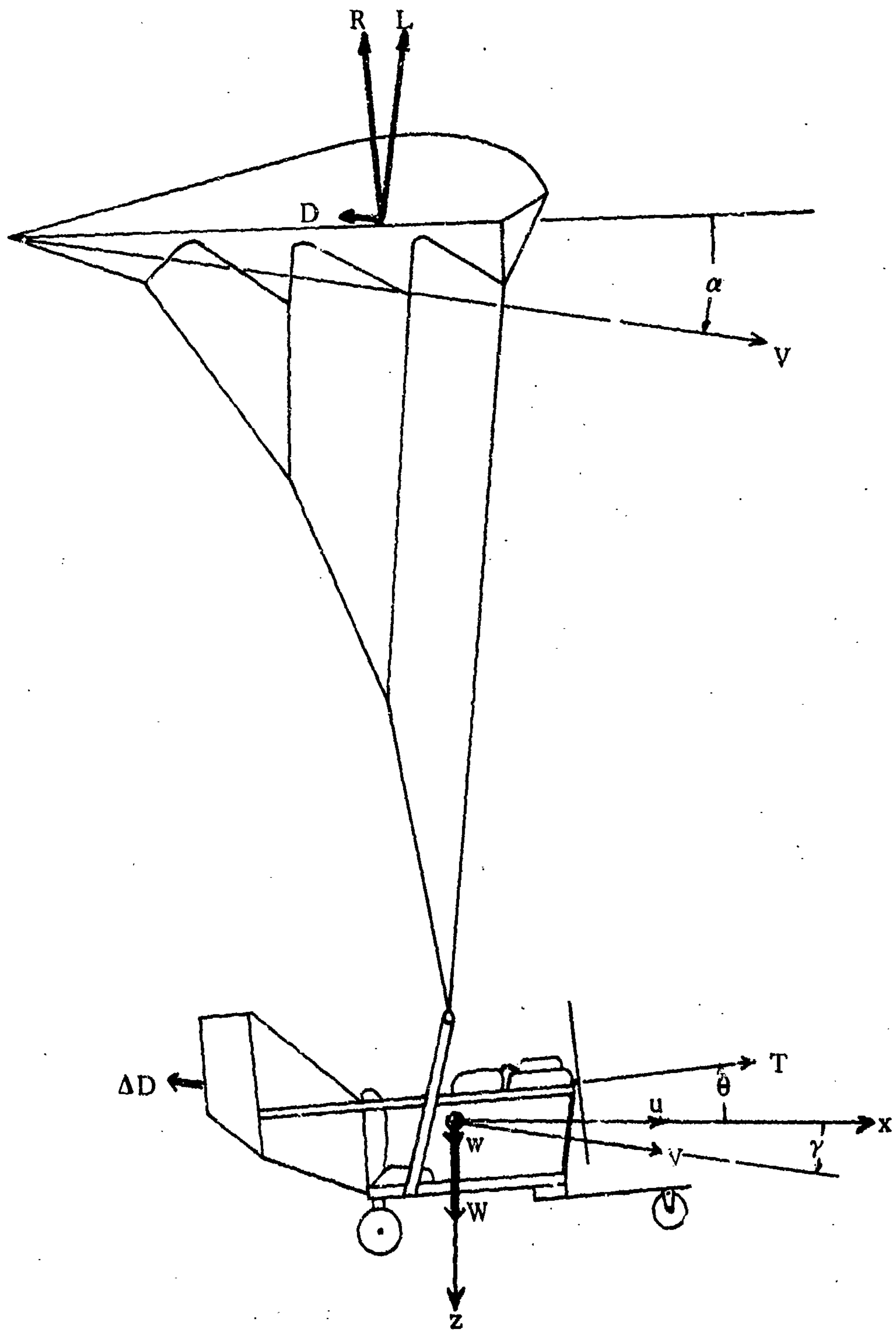


Figure 5. First Powered Parafoil Flight



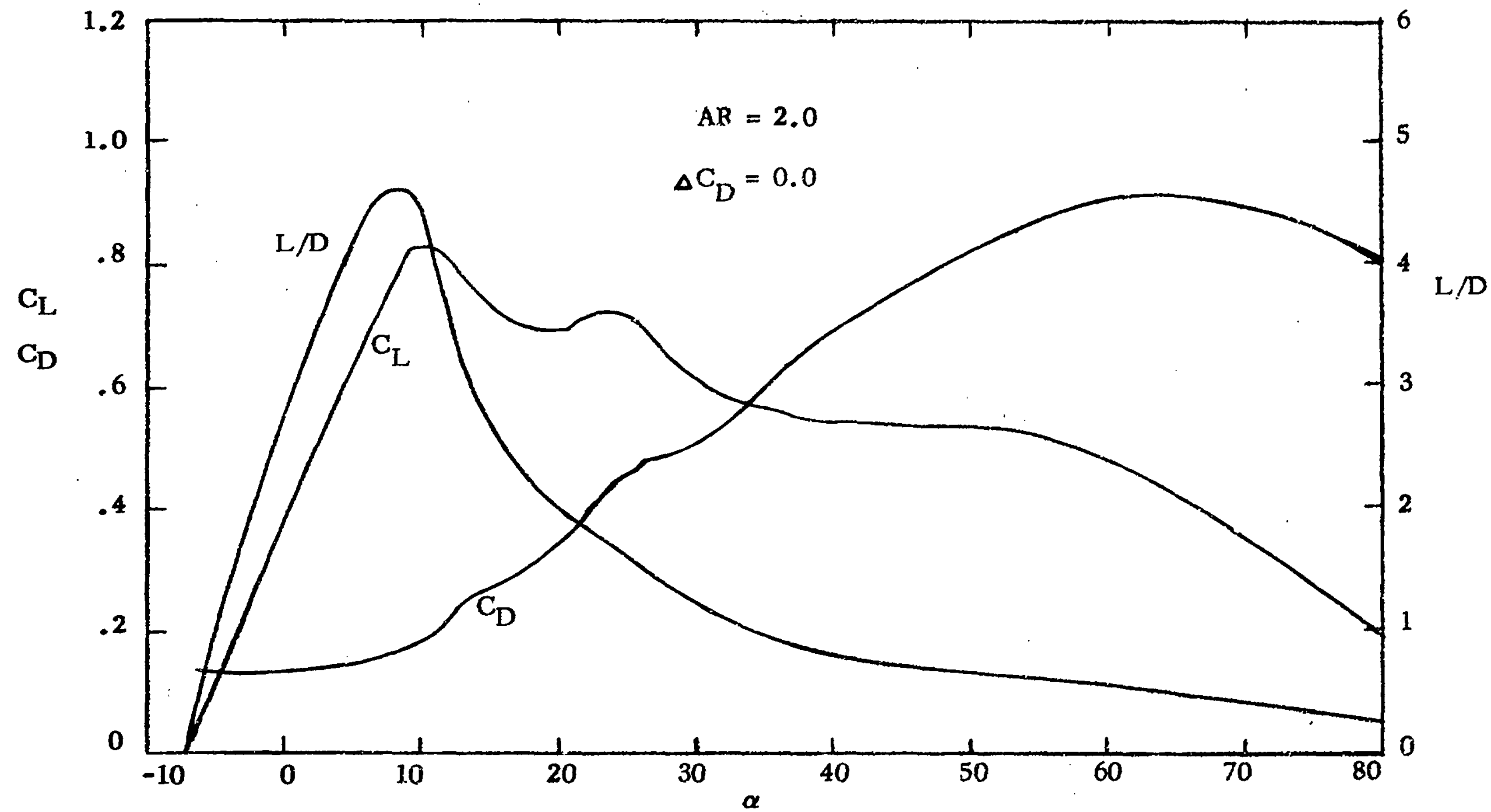
LEVEL FLIGHT

Figure 6a



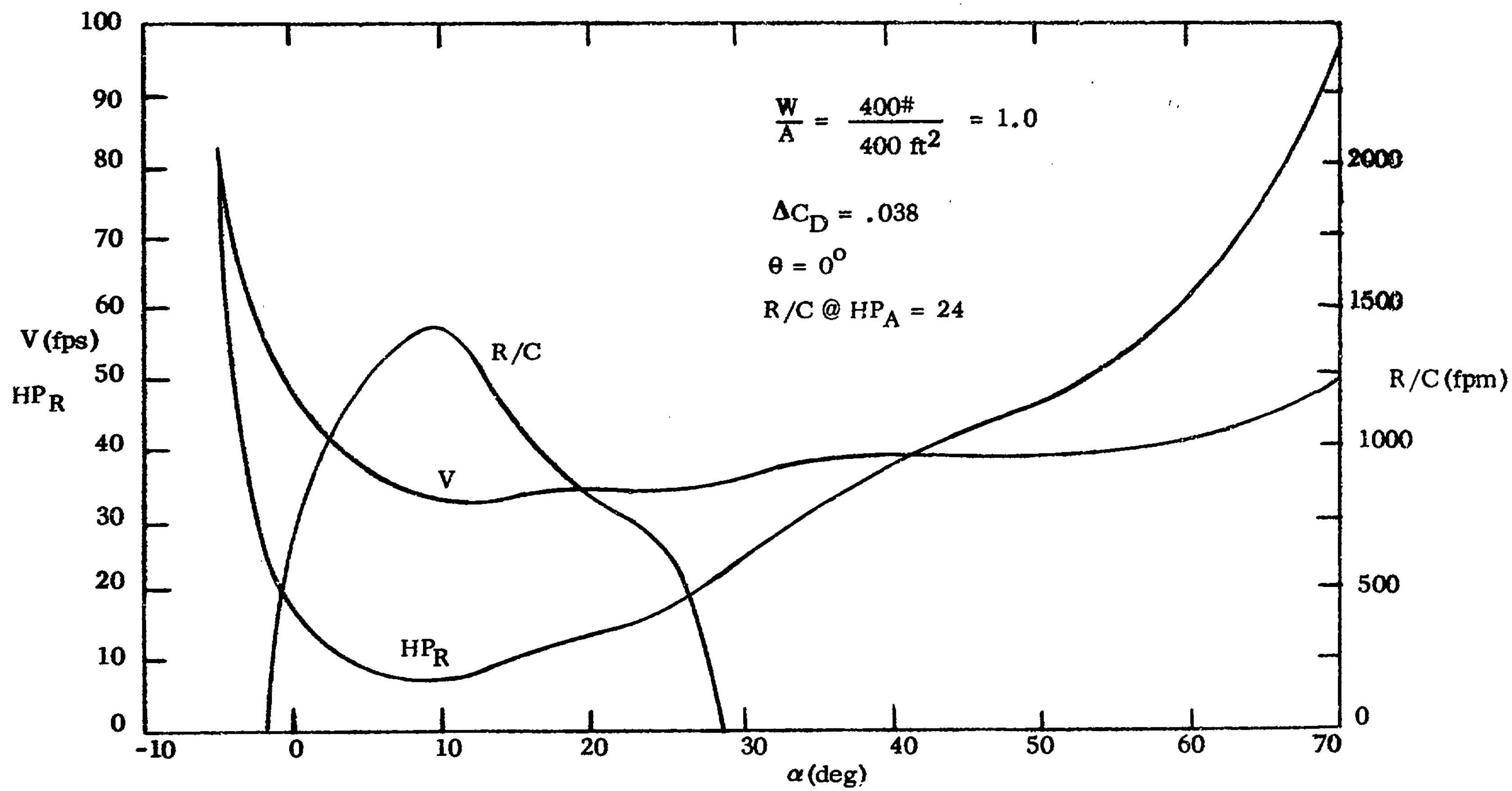
CLIMBING AND DESCENDING FLIGHT

Figure 6b



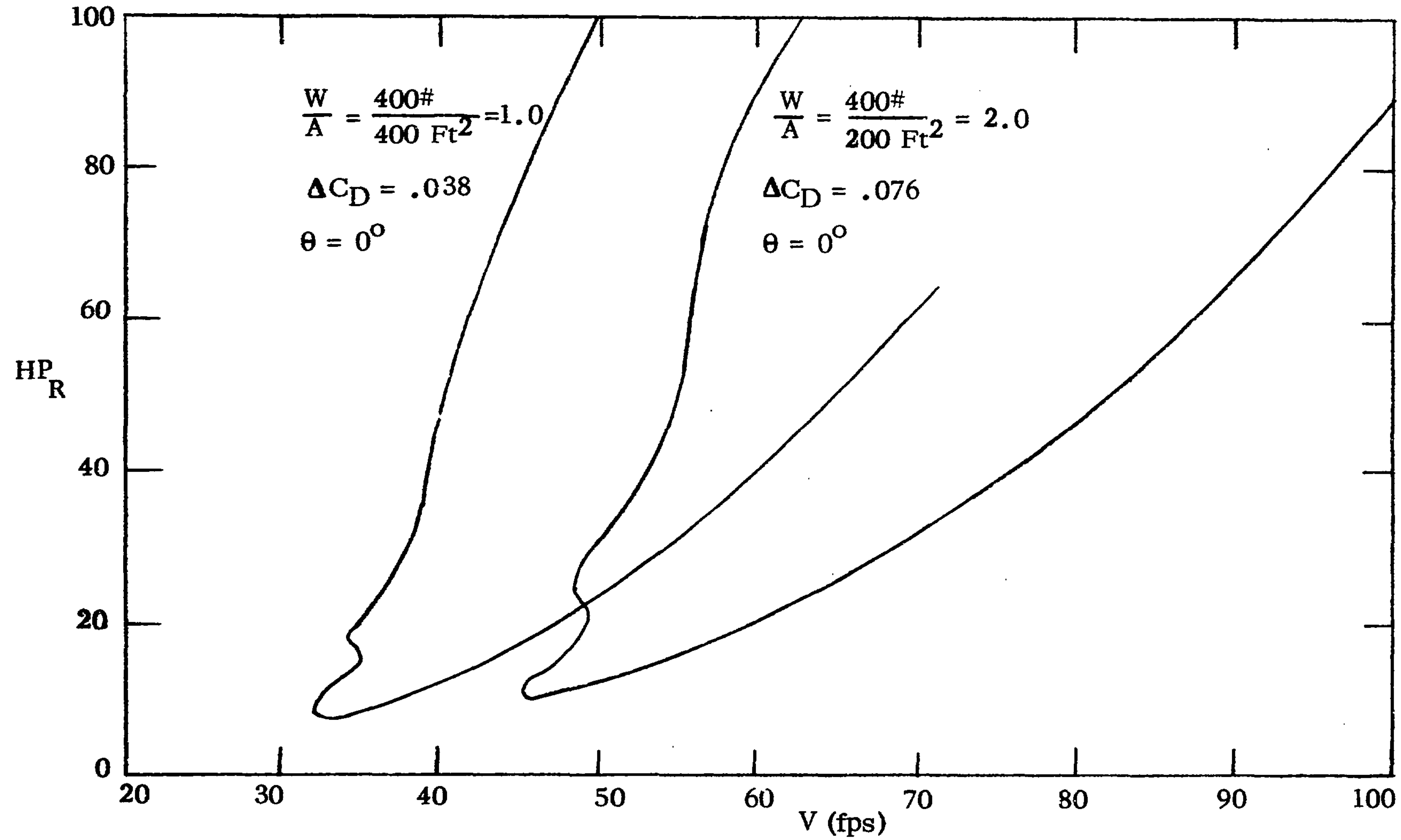
BASIC AERODYNAMICS OF PARAFOIL

Figure 7



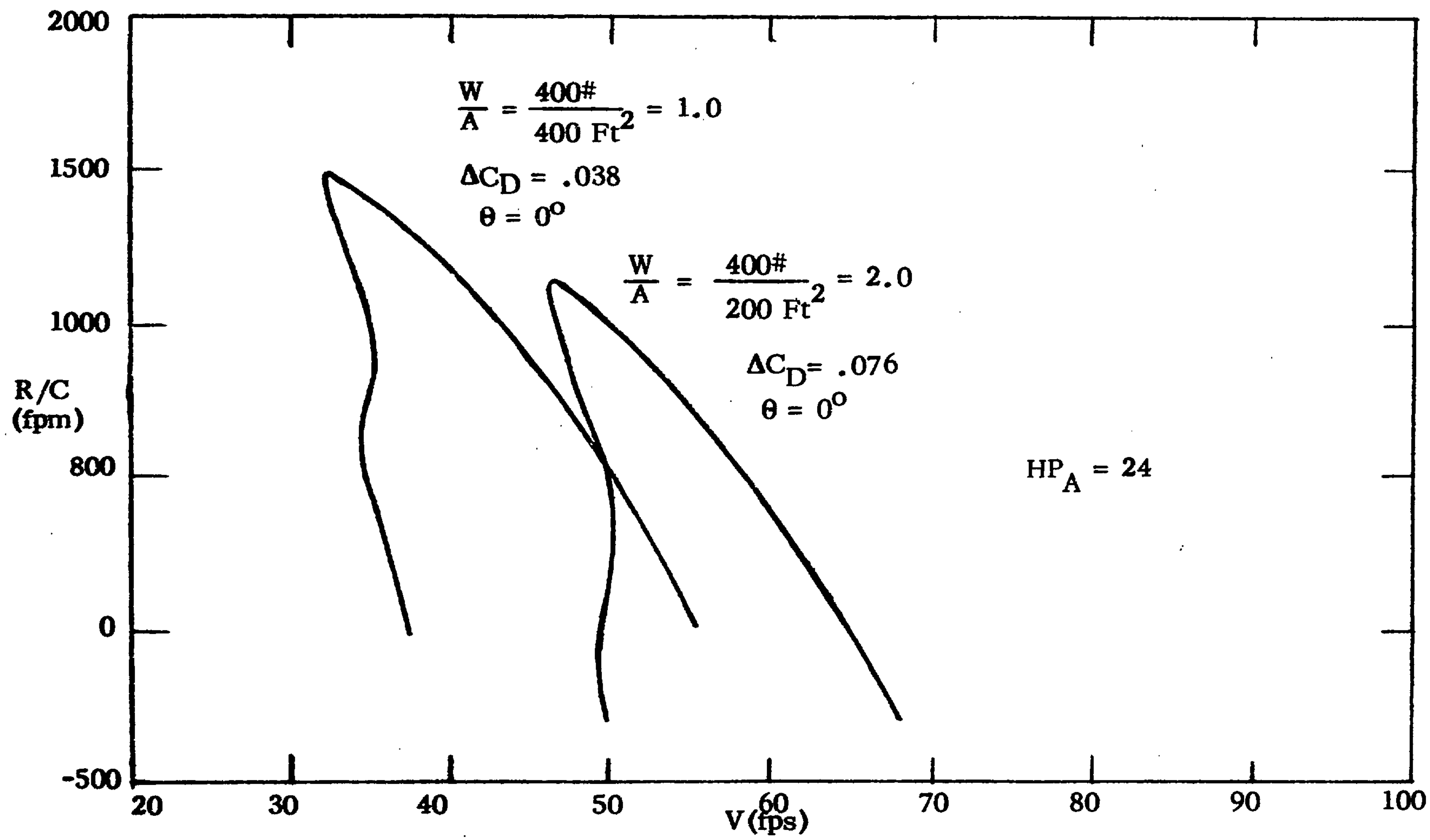
LEVEL FLIGHT PERFORMANCE

Figure 8



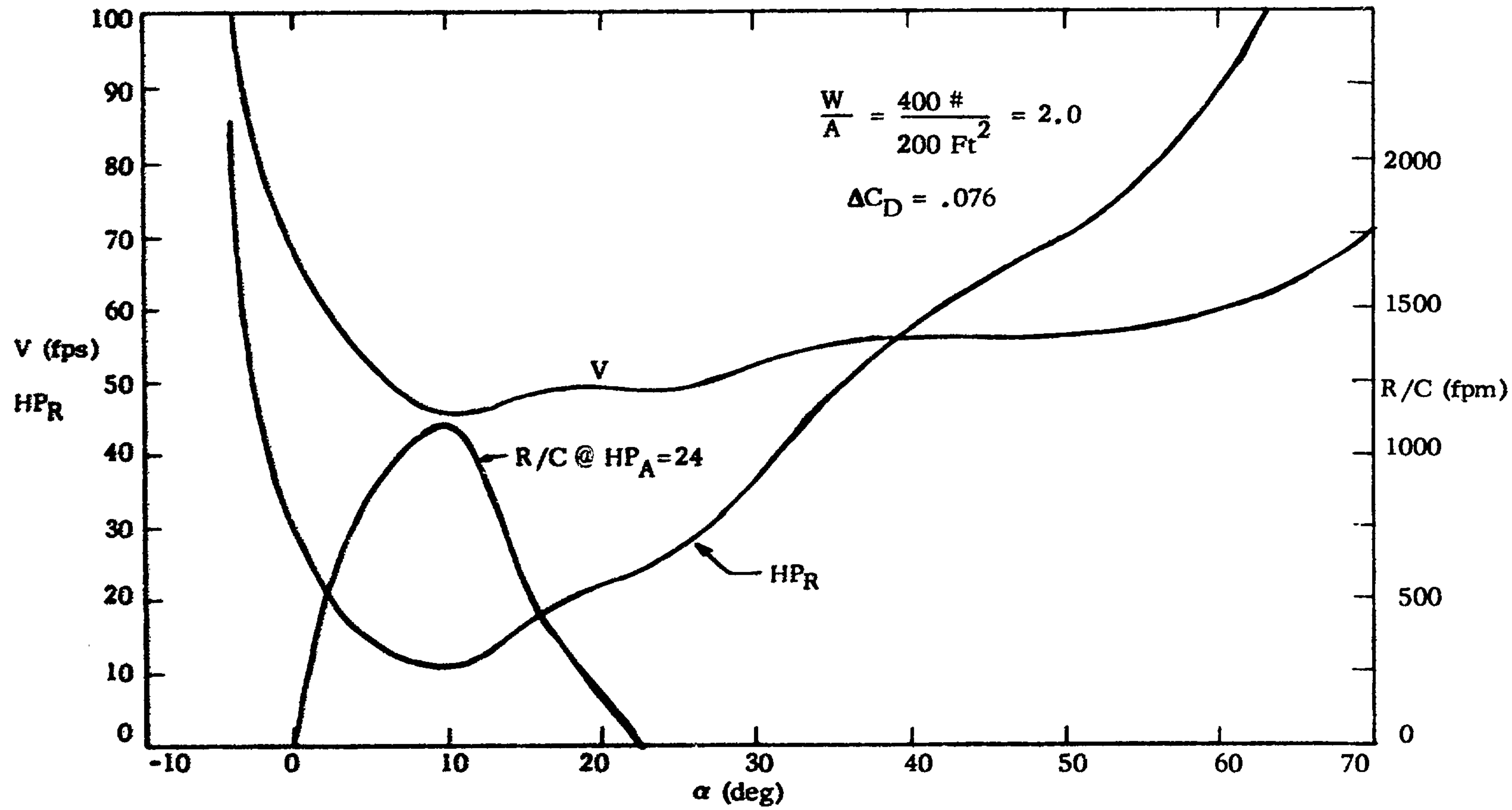
HORSEPOWER REQUIRED FOR LEVEL FLIGHT

Figure 9a



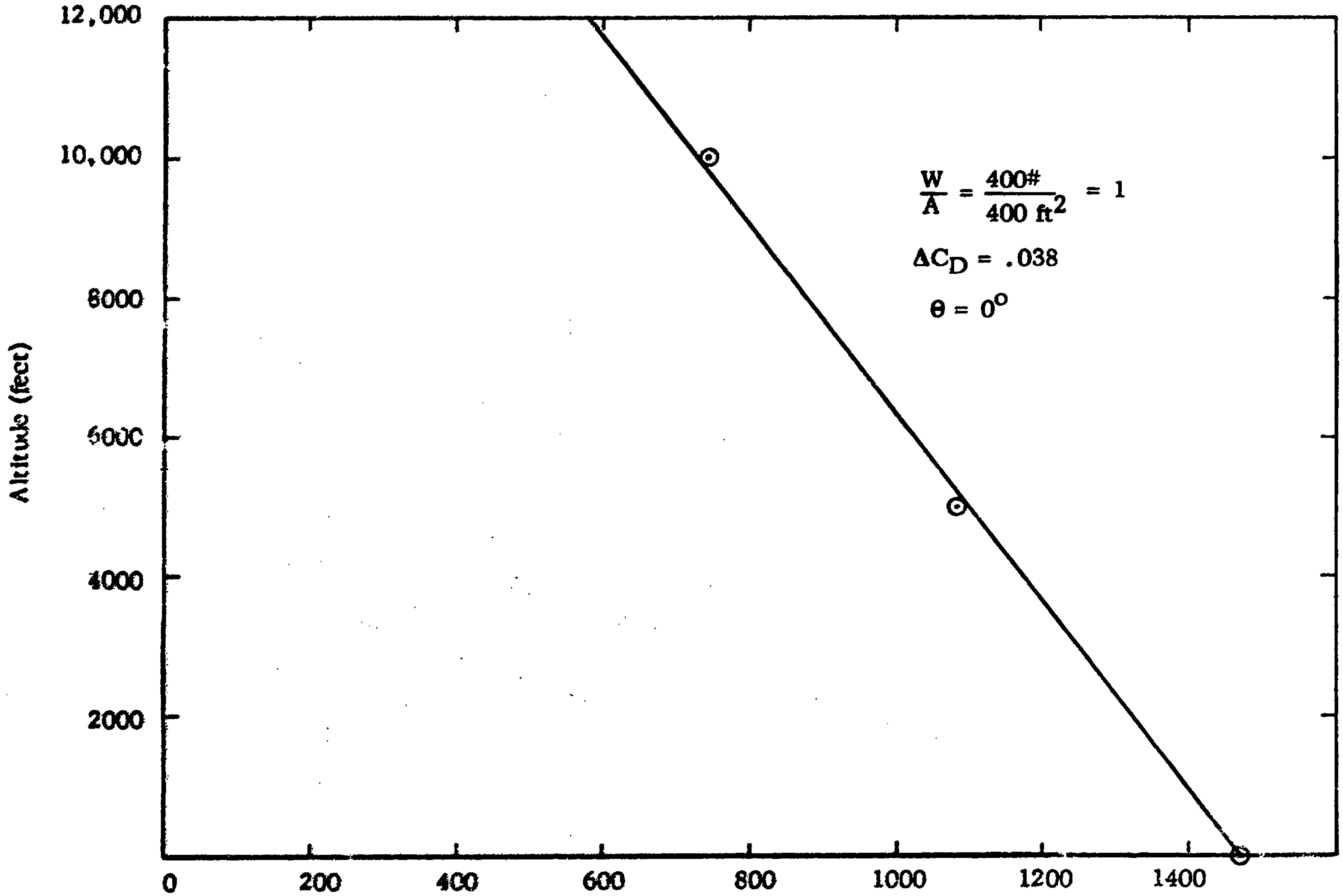
MAXIMUM RATE OF CLIMB AVAILABLE IN LEVEL FLIGHT

Figure 9b



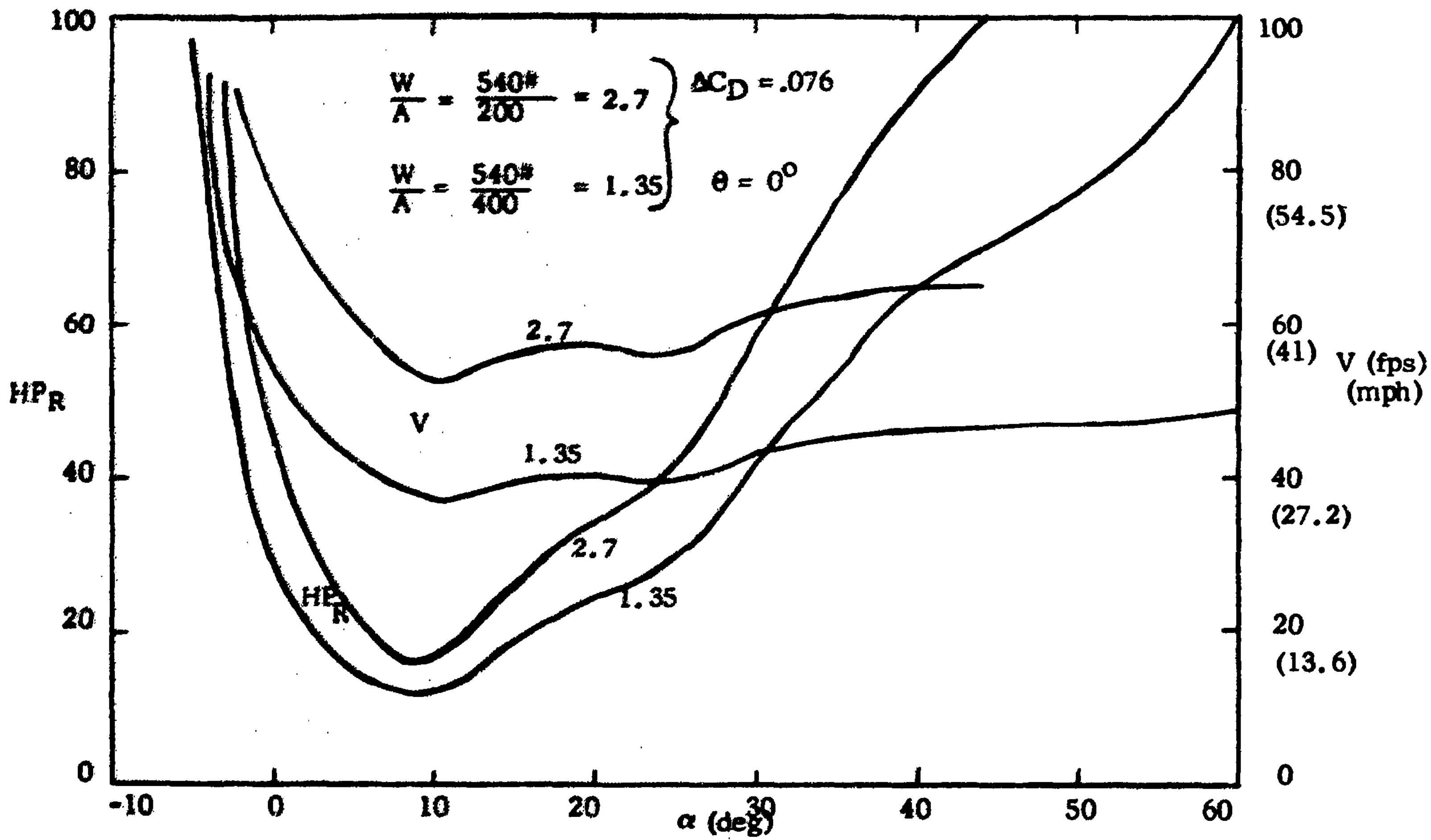
LEVEL FLIGHT PERFORMANCE

Figure 10



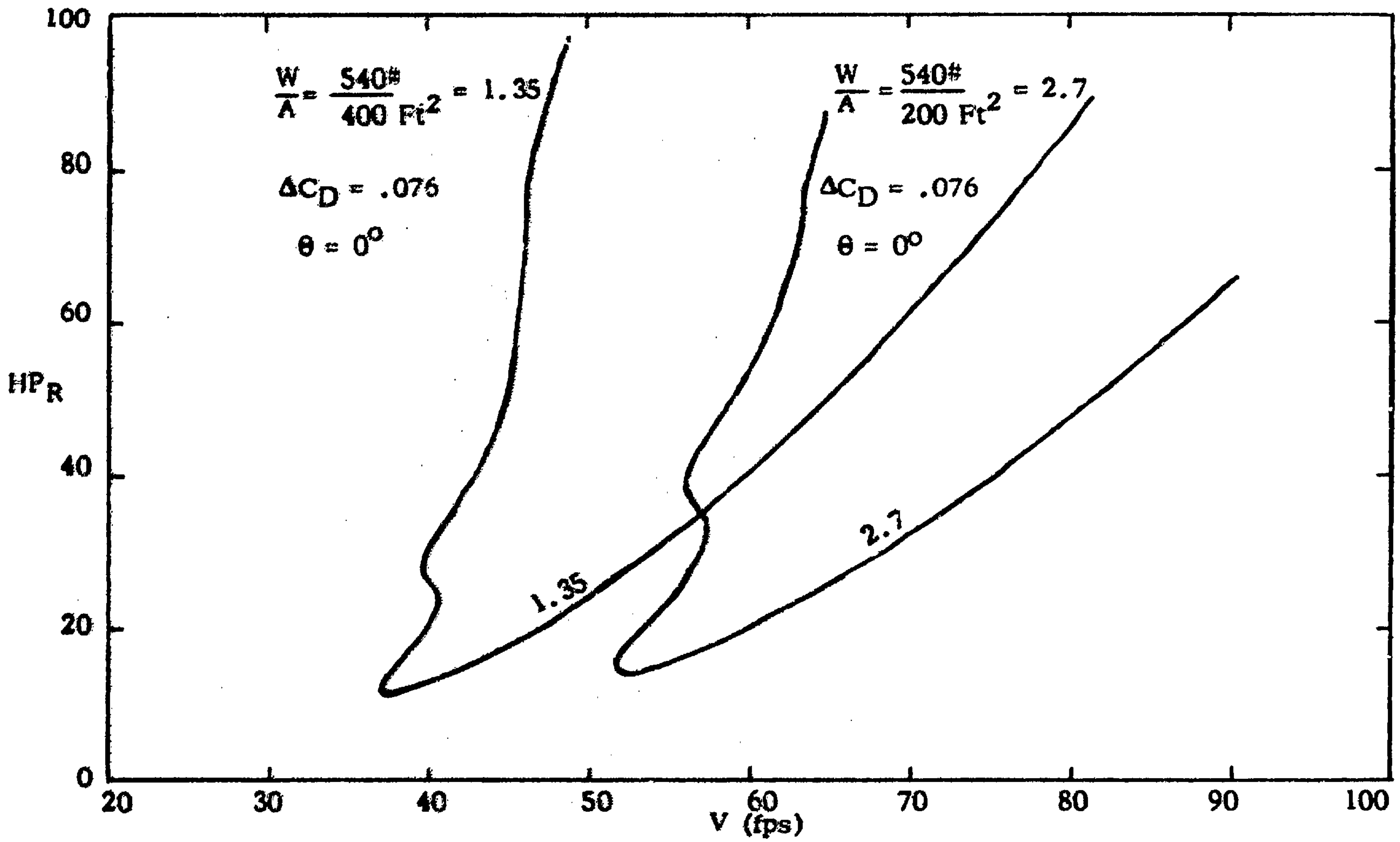
MAXIMUM RATE OF CLIMB AVAILABLE VERSUS ALTITUDE

Figure 11



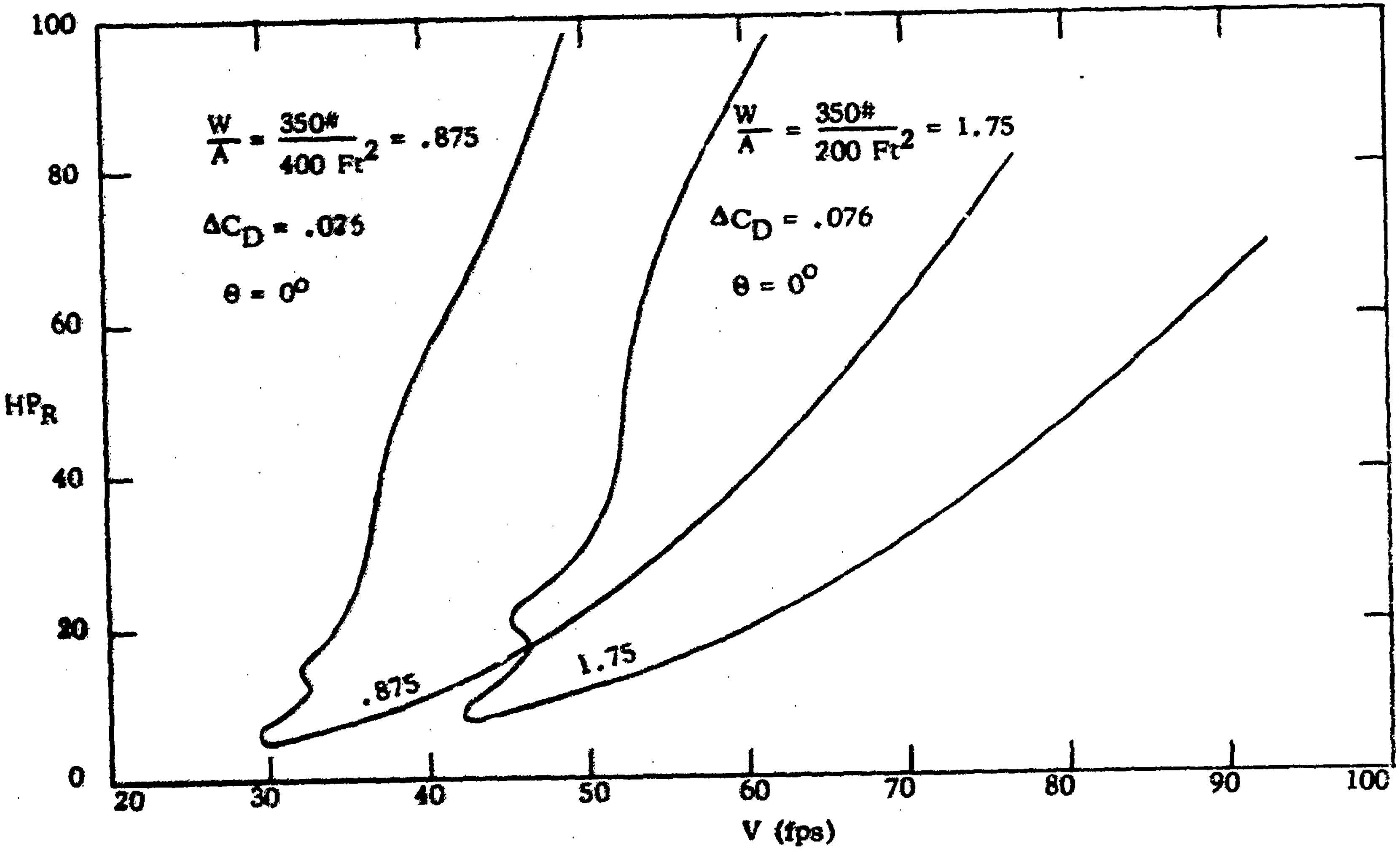
IRISH FLYER LEVEL FLIGHT PERFORMANCE

Figure 12



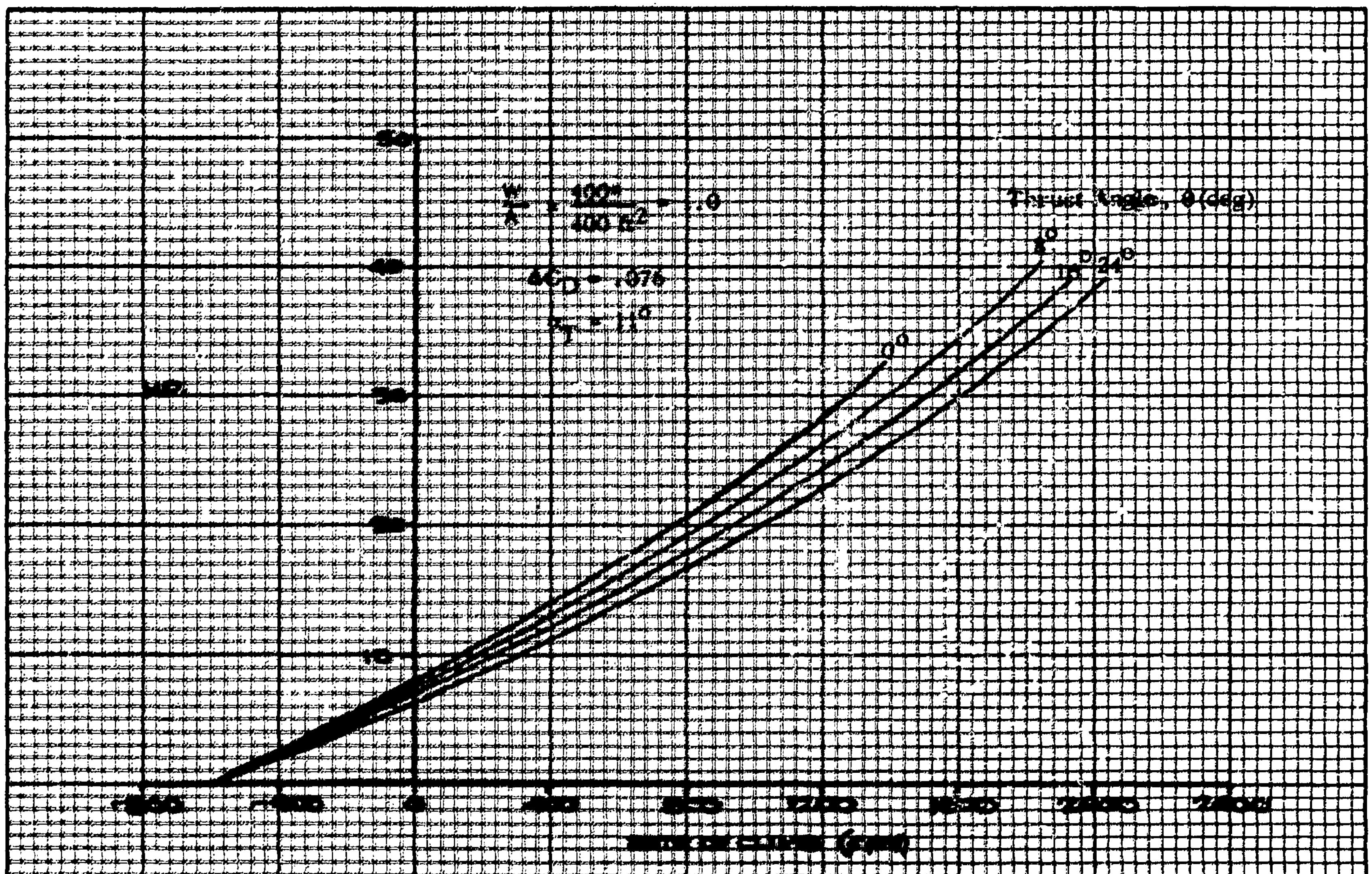
IRISH FLYER HORSEPOWER REQUIRED FOR LEVEL FLIGHT

Figure 13



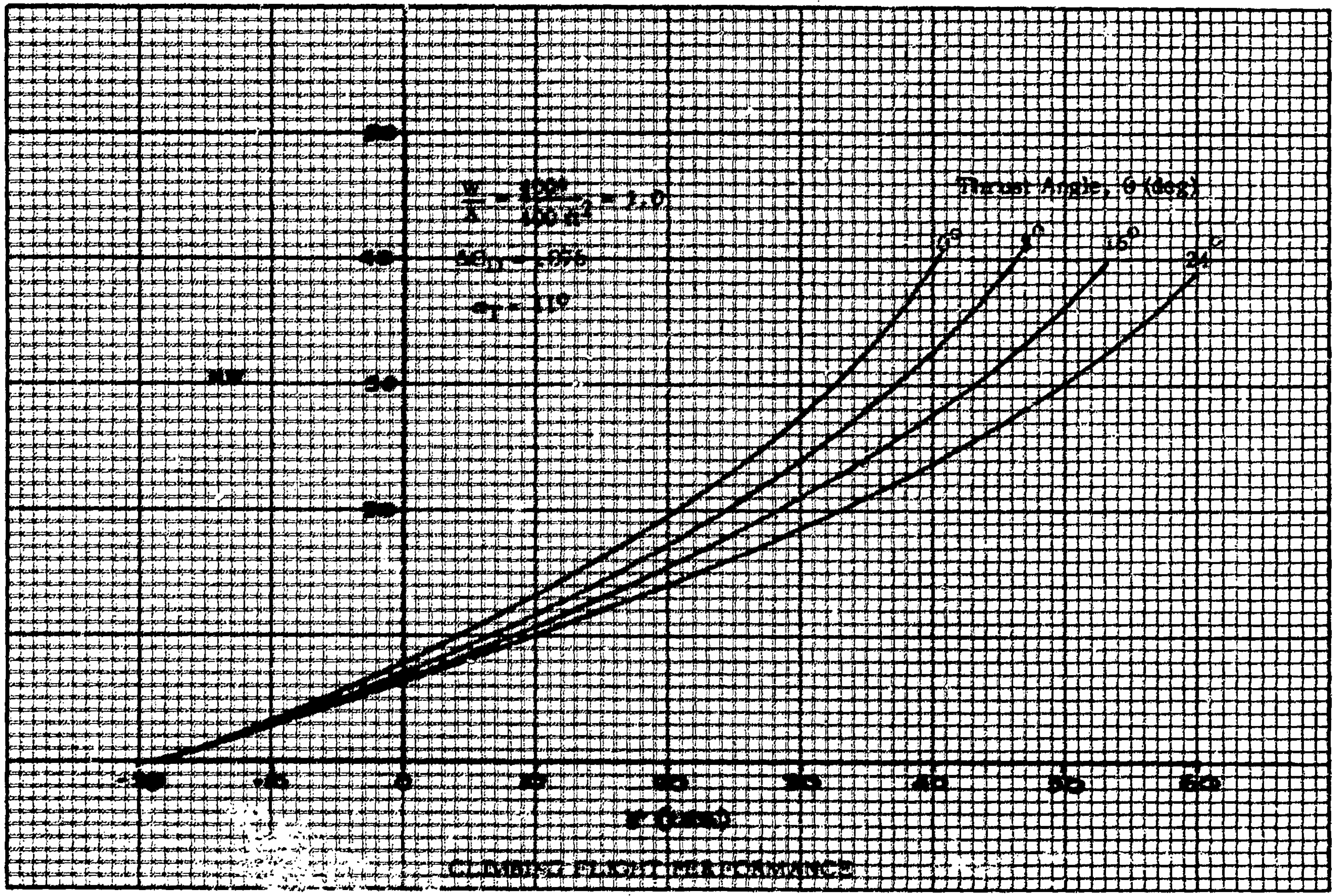
IRISH FLYER HORSEPOWER REQUIRED FOR LEVEL FLIGHT

Figure 14



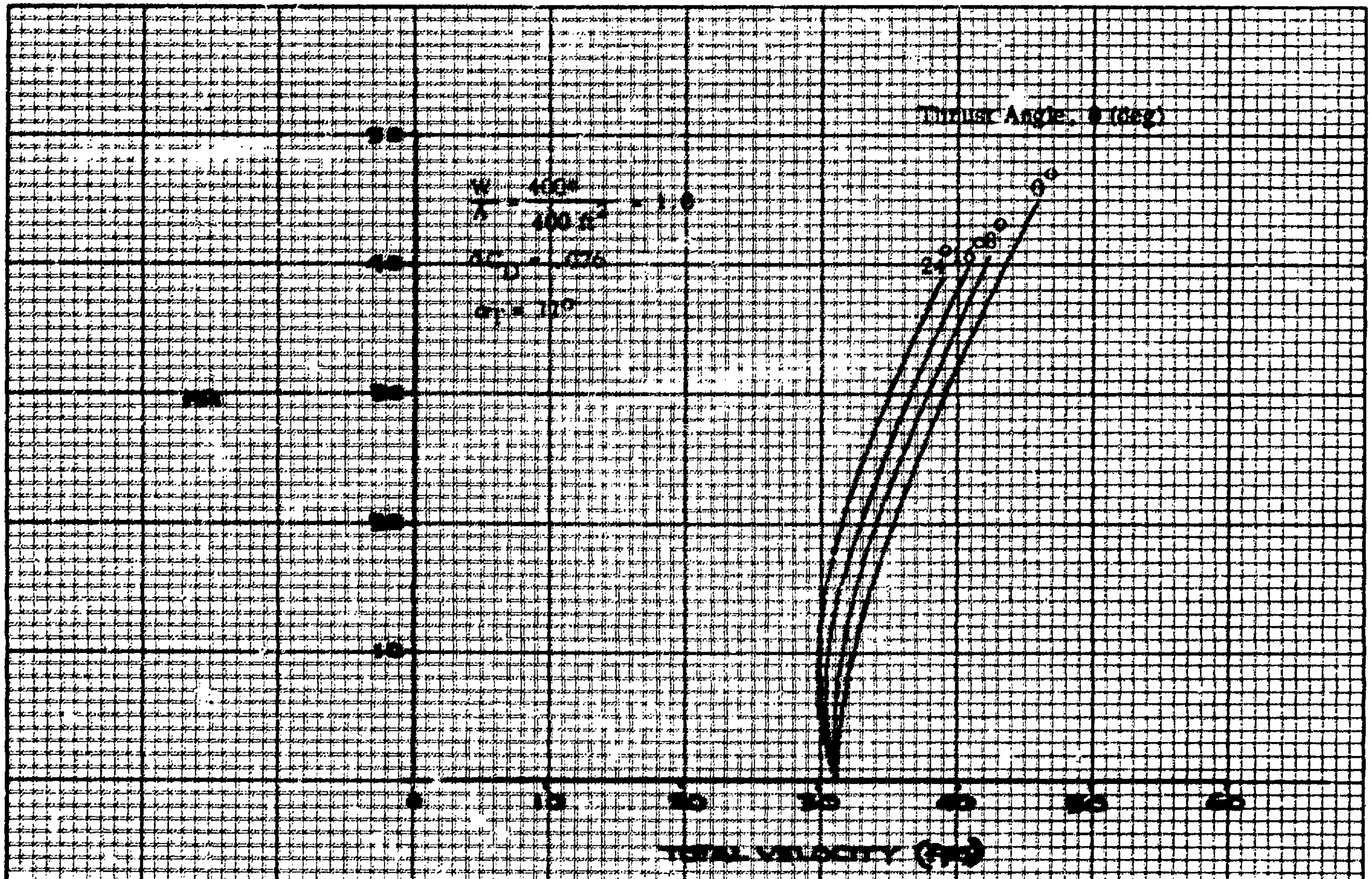
CLIMBING FLIGHT PERFORMANCE

Figure 15a



CLIMBING FLIGHT PERFORMANCE

Figure 15b



CLIMBING FLIGHT PERFORMANCE

Figure 15c

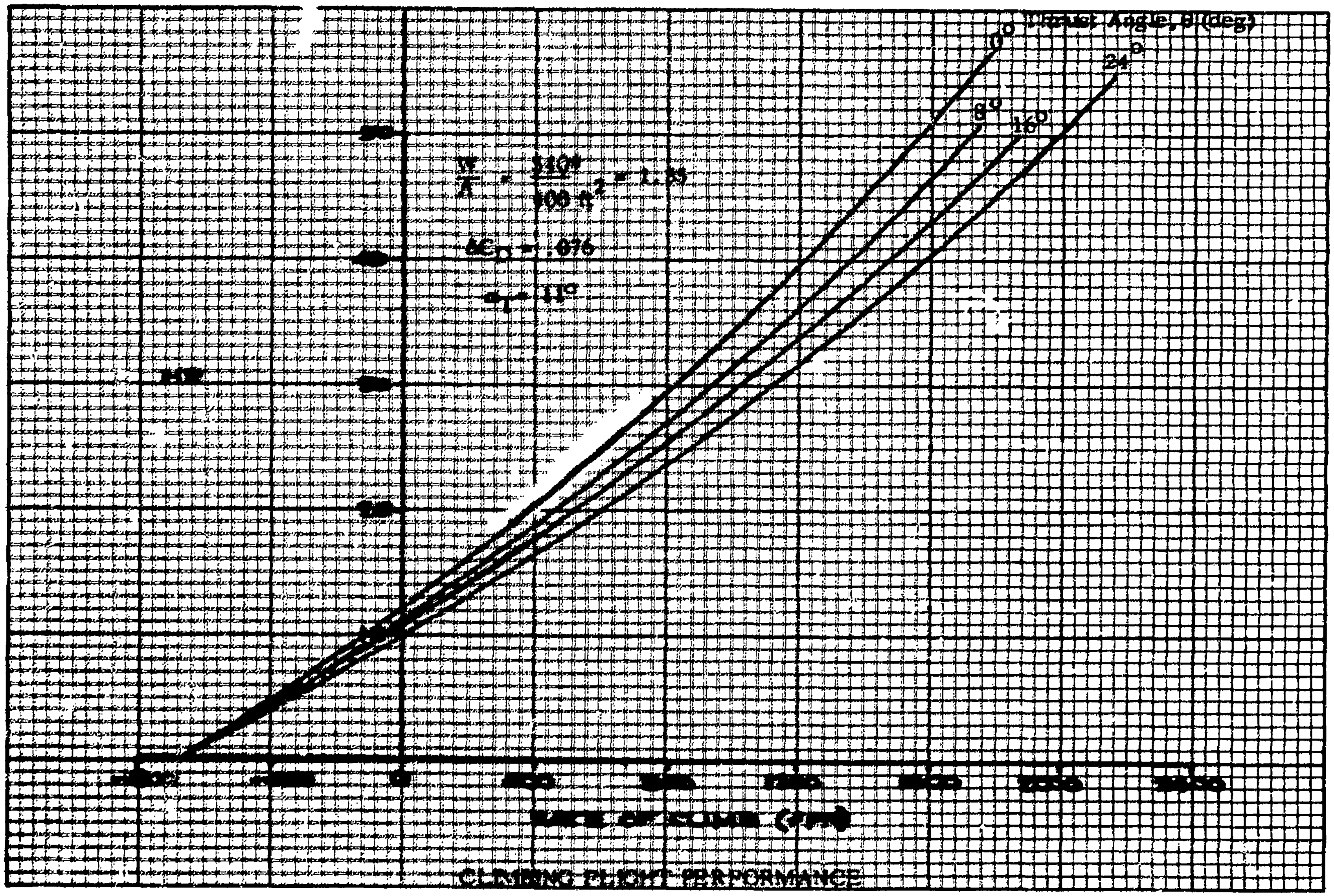
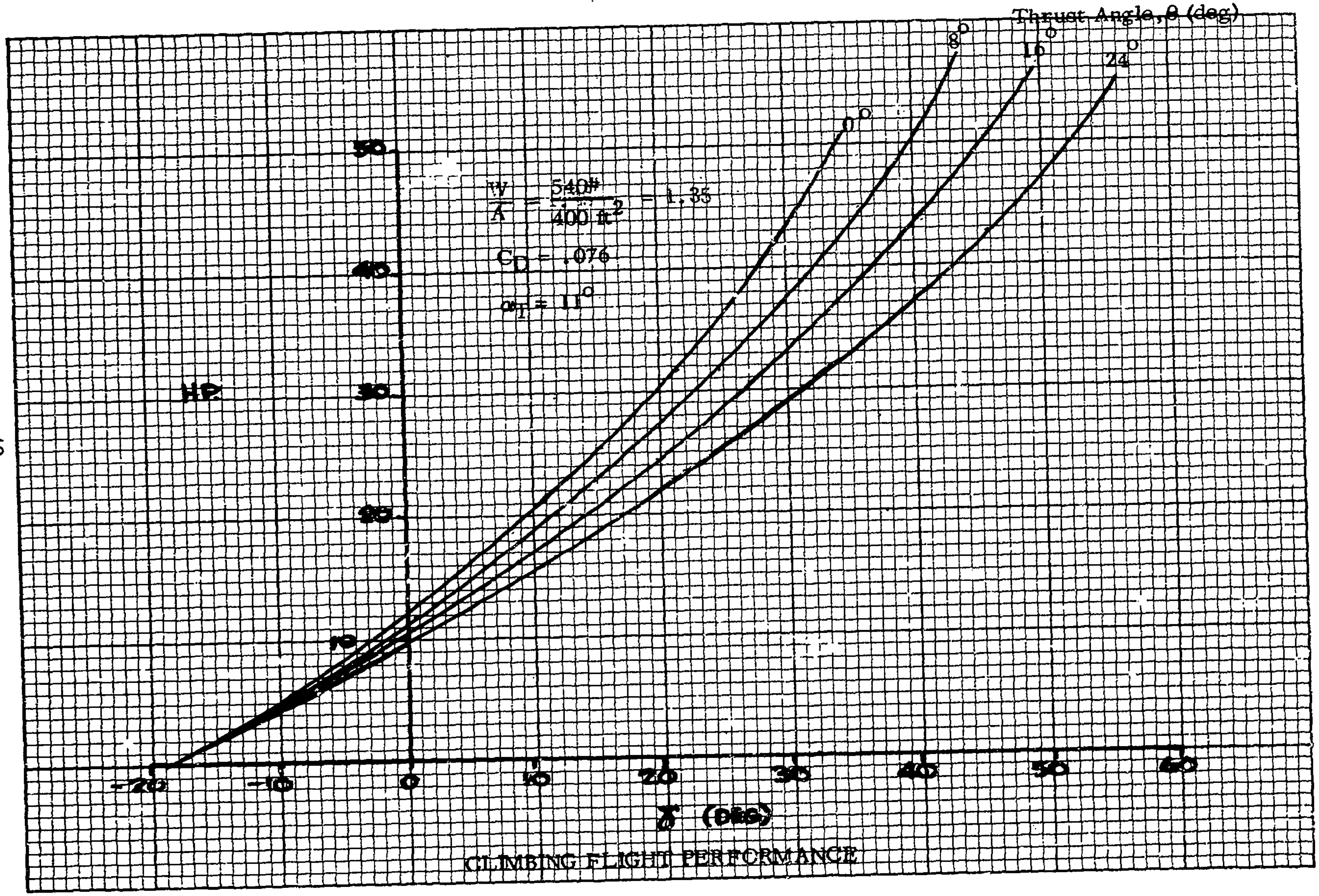
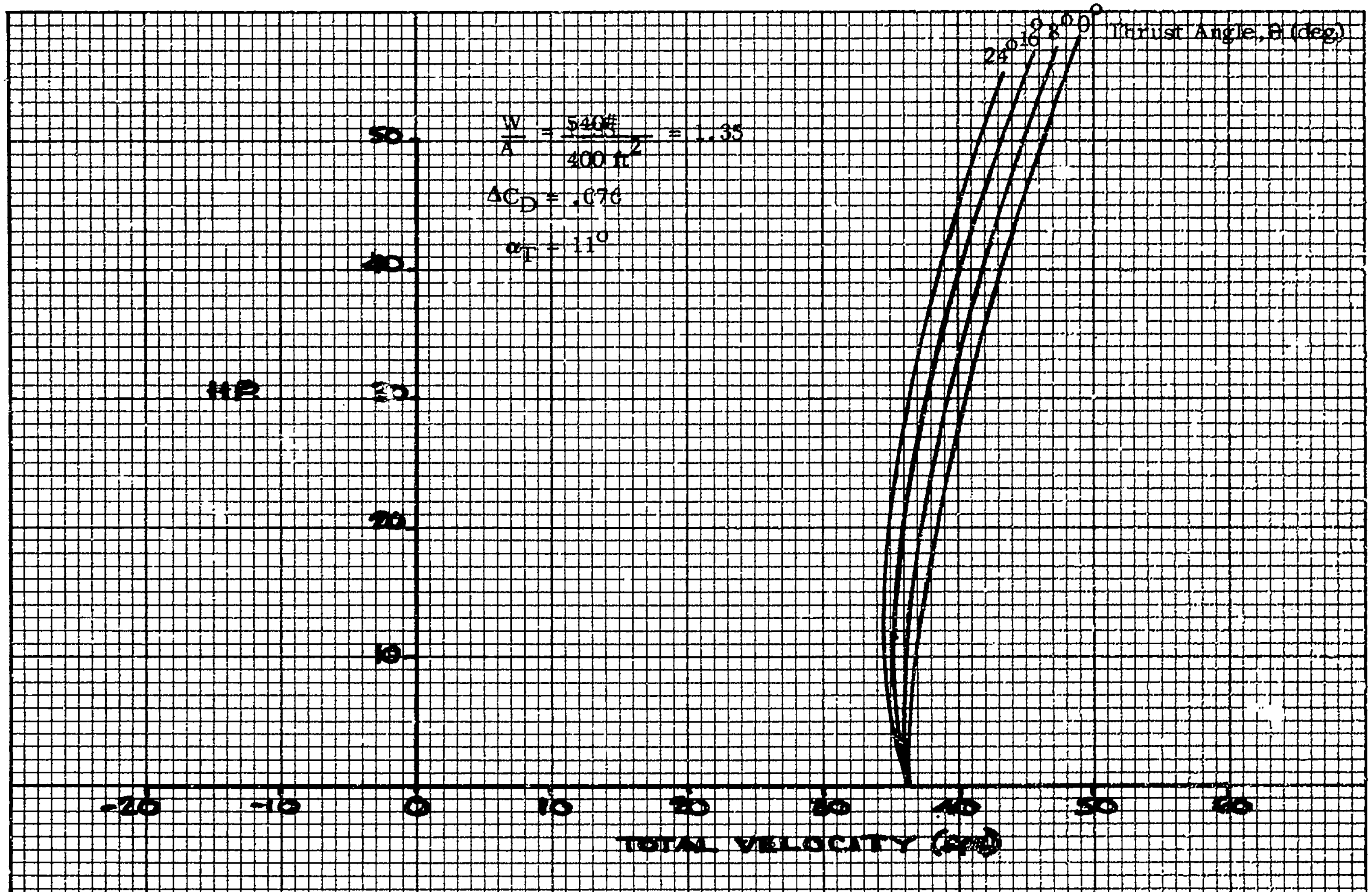


Figure 16a



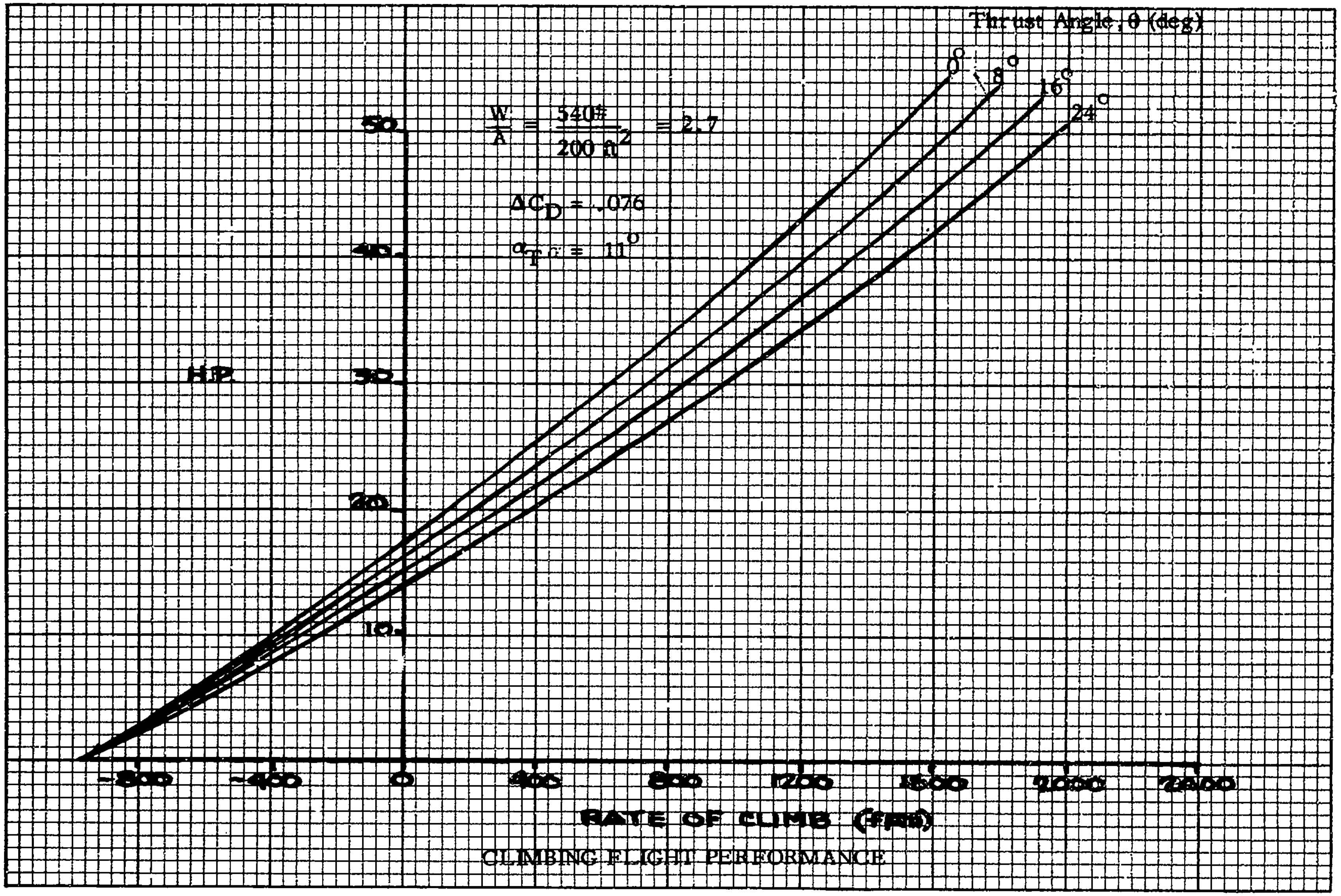
CLIMBING FLIGHT PERFORMANCE

Figure 16b



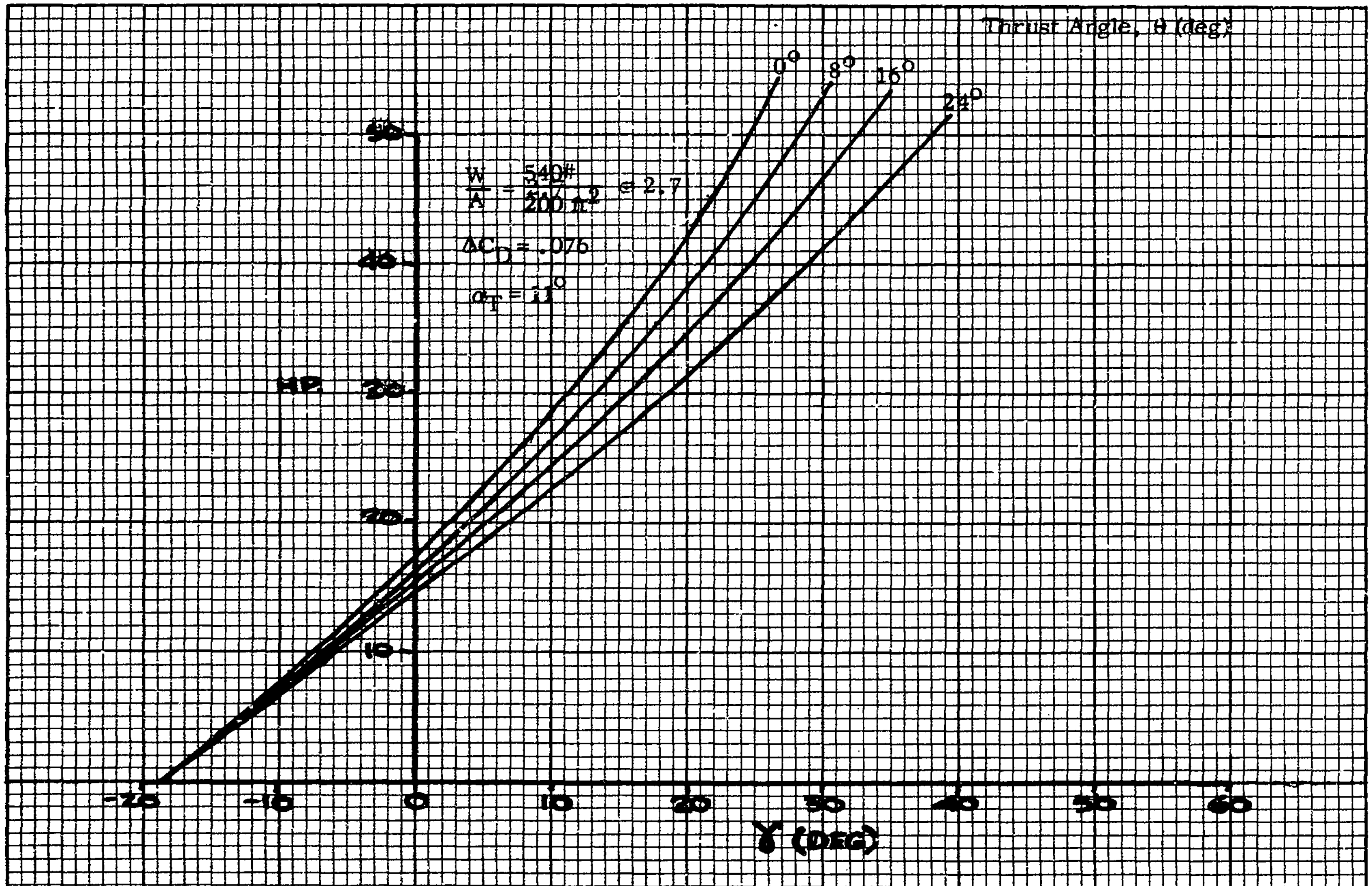
CLIMBING FLIGHT PERFORMANCE

Figure 16c



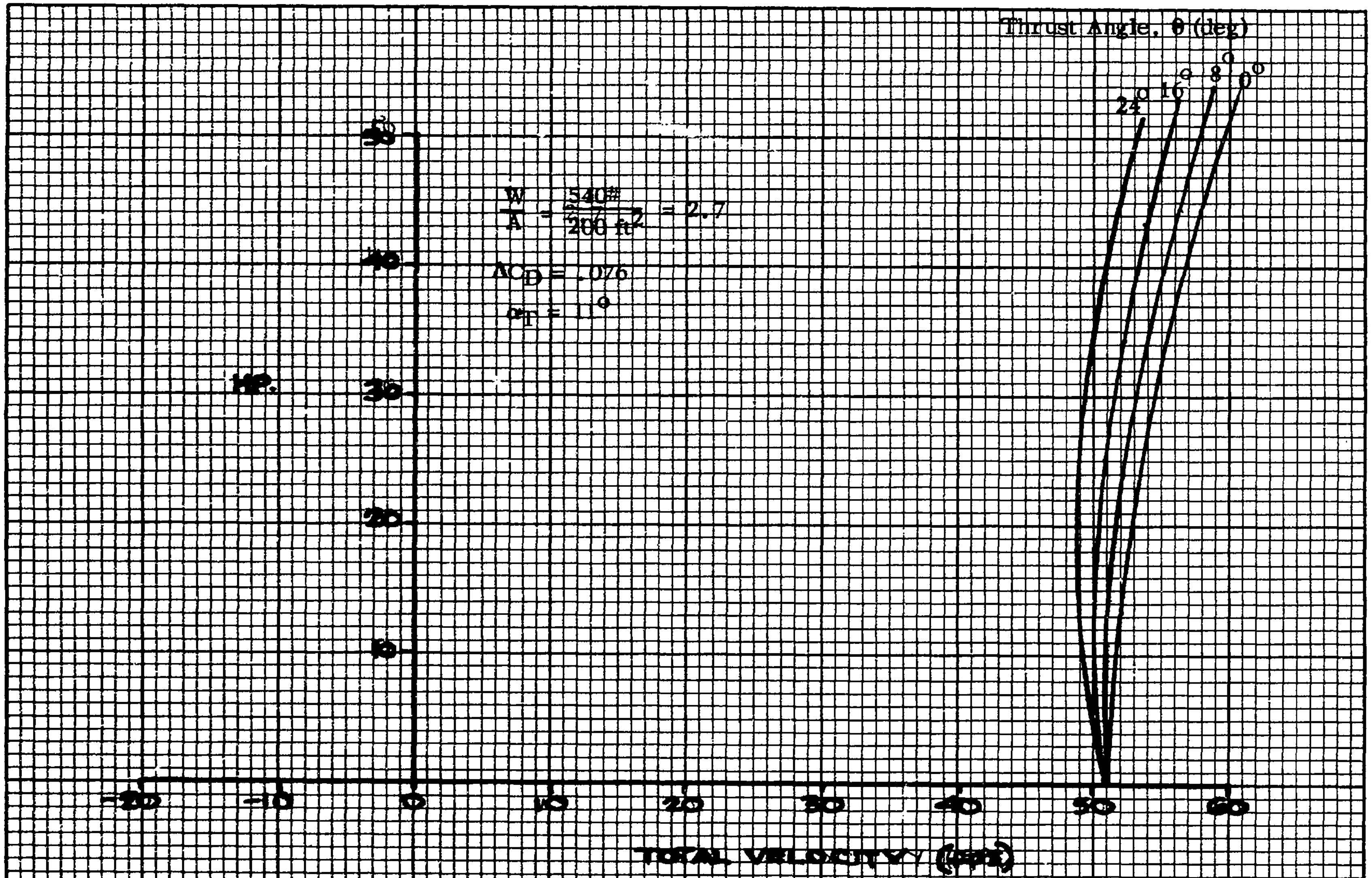
Thrust Angle, θ (deg)
0°
8°
16°
24°
$\frac{W}{A} = \frac{540\#}{200\ ft^2} = 2.7$
$\Delta C_D = .076$
$\alpha_{T} = 11^{\circ}$
H.P.
50
40
30
20
10
-800
-400
0
400
800
1200
1600
2000
2400
RATE OF CLIMB (fpm)
CLIMBING FLIGHT PERFORMANCE

Figure 17a



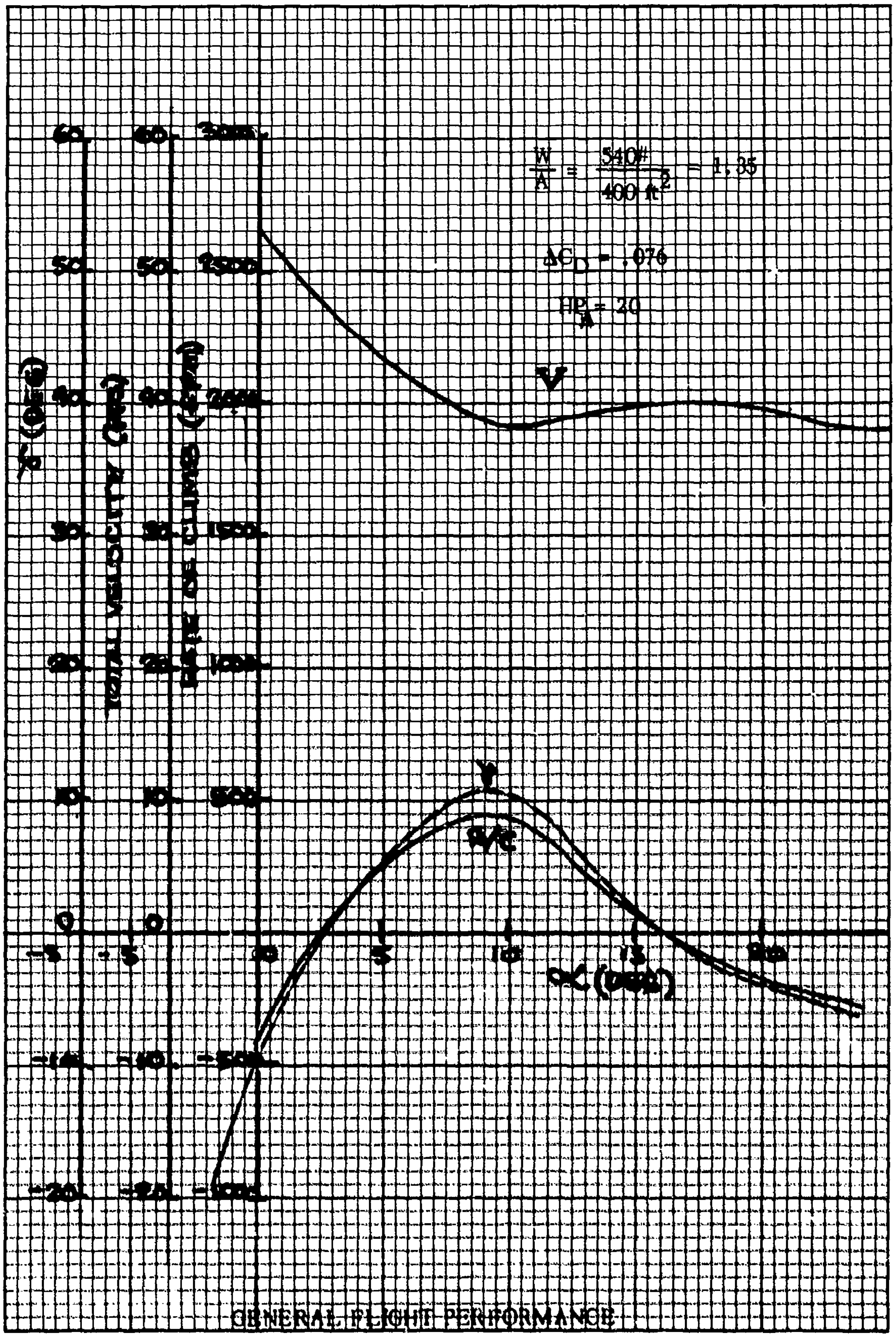
CLIMBING FLIGHT PERFORMANCE

Figure 17b



CLIMBING FLIGHT PERFORMANCE

Figure 17c



$\frac{W}{A} = \frac{540\#}{400\ ft^2} = 1.35$
$\Delta C_D = .076$
$HP_A = 20$
V
R/C
α (DEG)
GENERAL FLIGHT PERFORMANCE

Figure 18a

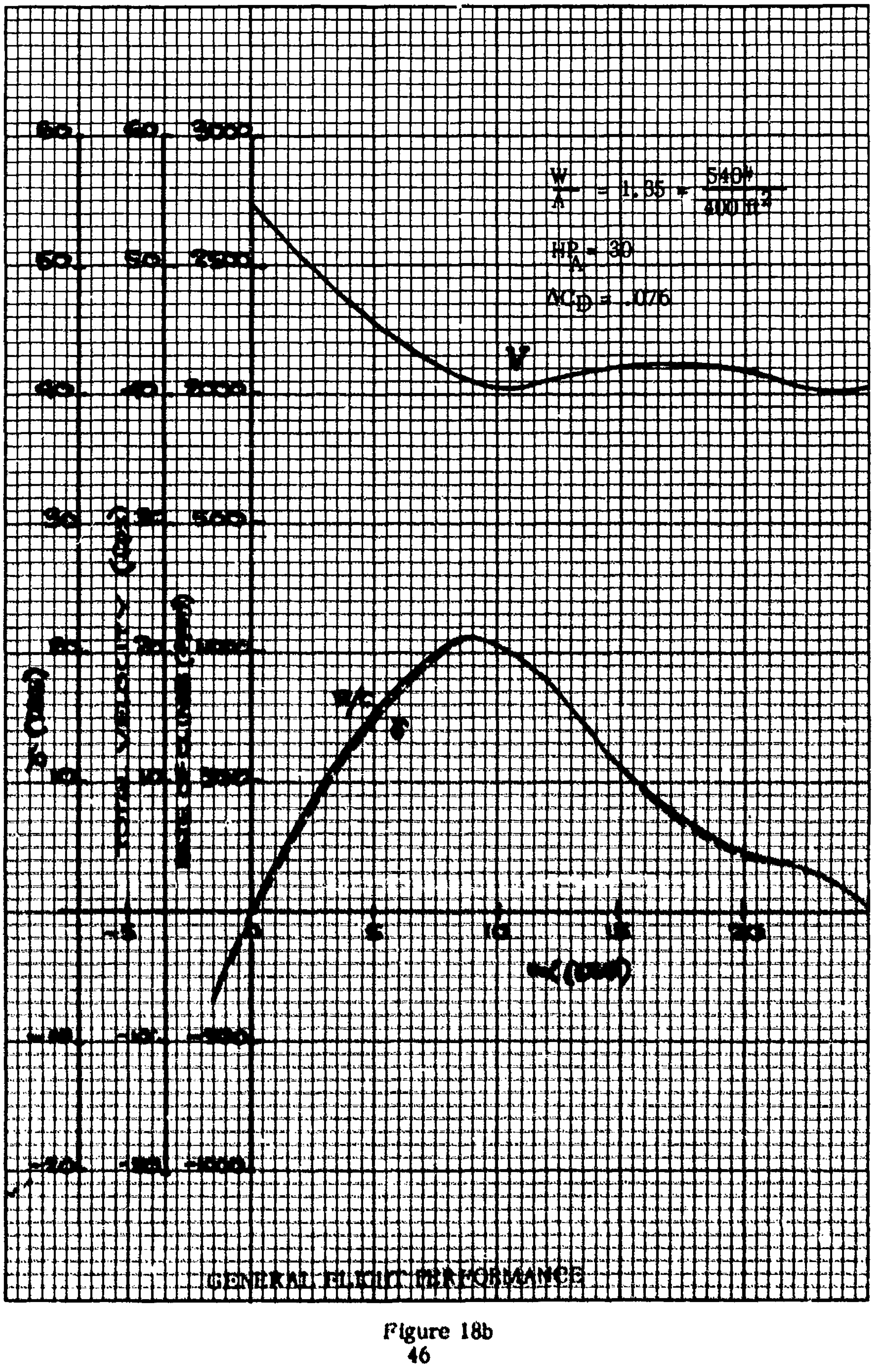
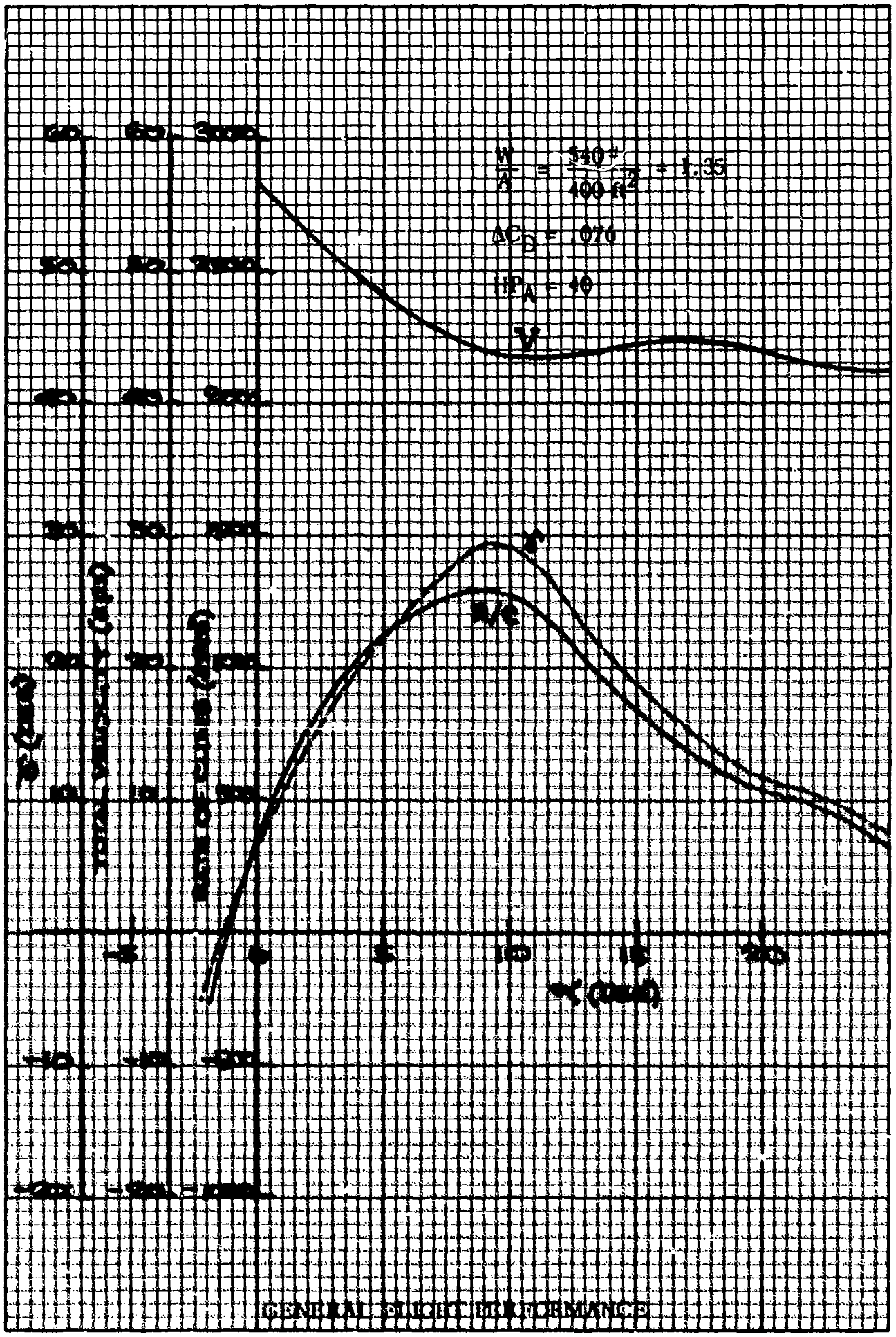
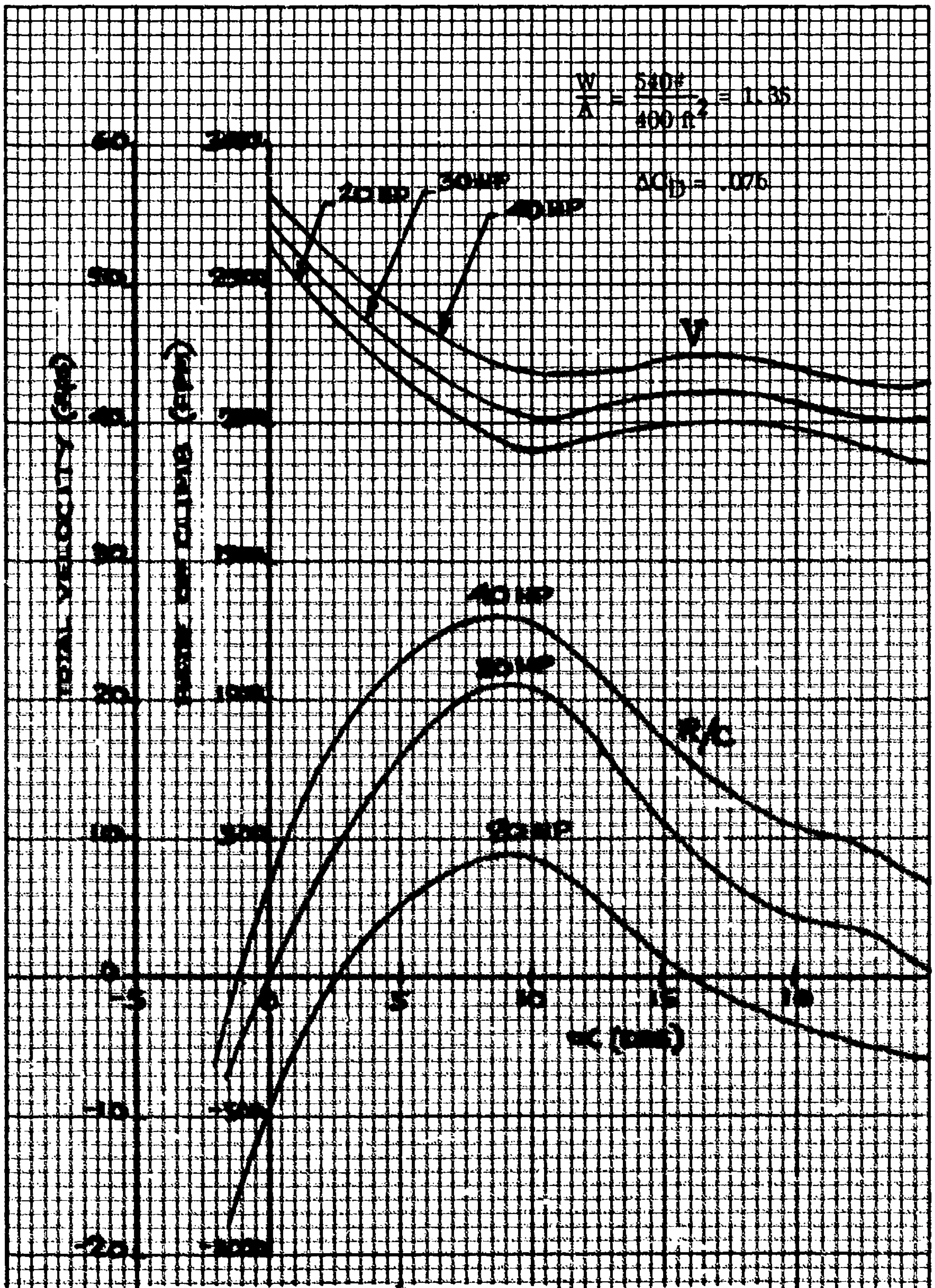

W/A = 1.35 = 540#/400 ft²
HP_A = 30
ΔC_D = .076
V
TOTAL VELOCITY
GENERAL FLIGHT PERFORMANCE

Figure 18b



$\frac{W}{A} = \frac{540\#}{400\ ft^2} = 1.35$
$\Delta C_D = .070$
$HP_A = 40$
V
R/C
GENERAL FLIGHT PERFORMANCE

Figure 18c



SUMMARY OF GENERAL FLIGHT PERFORMANCE

Figure [illegible]

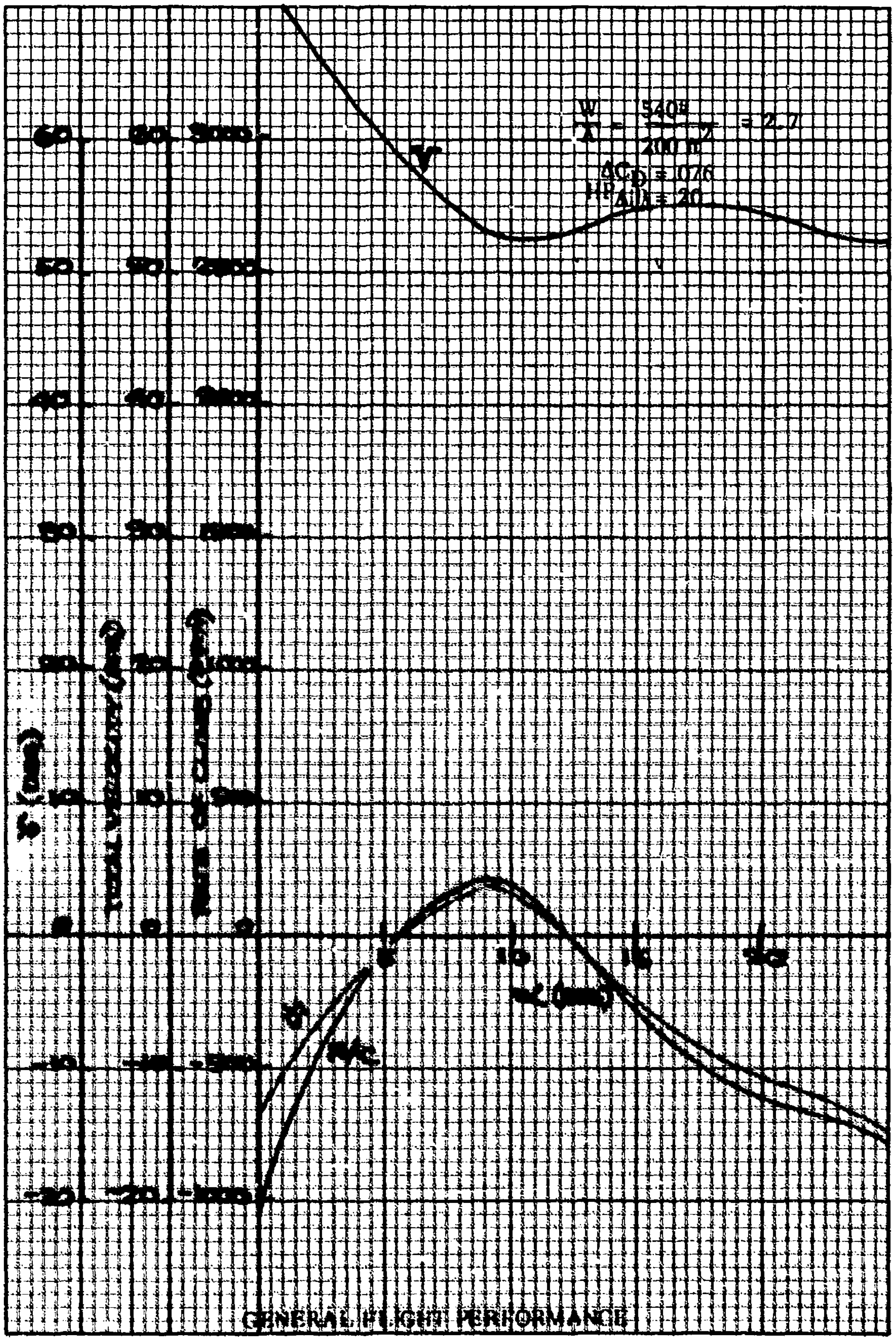
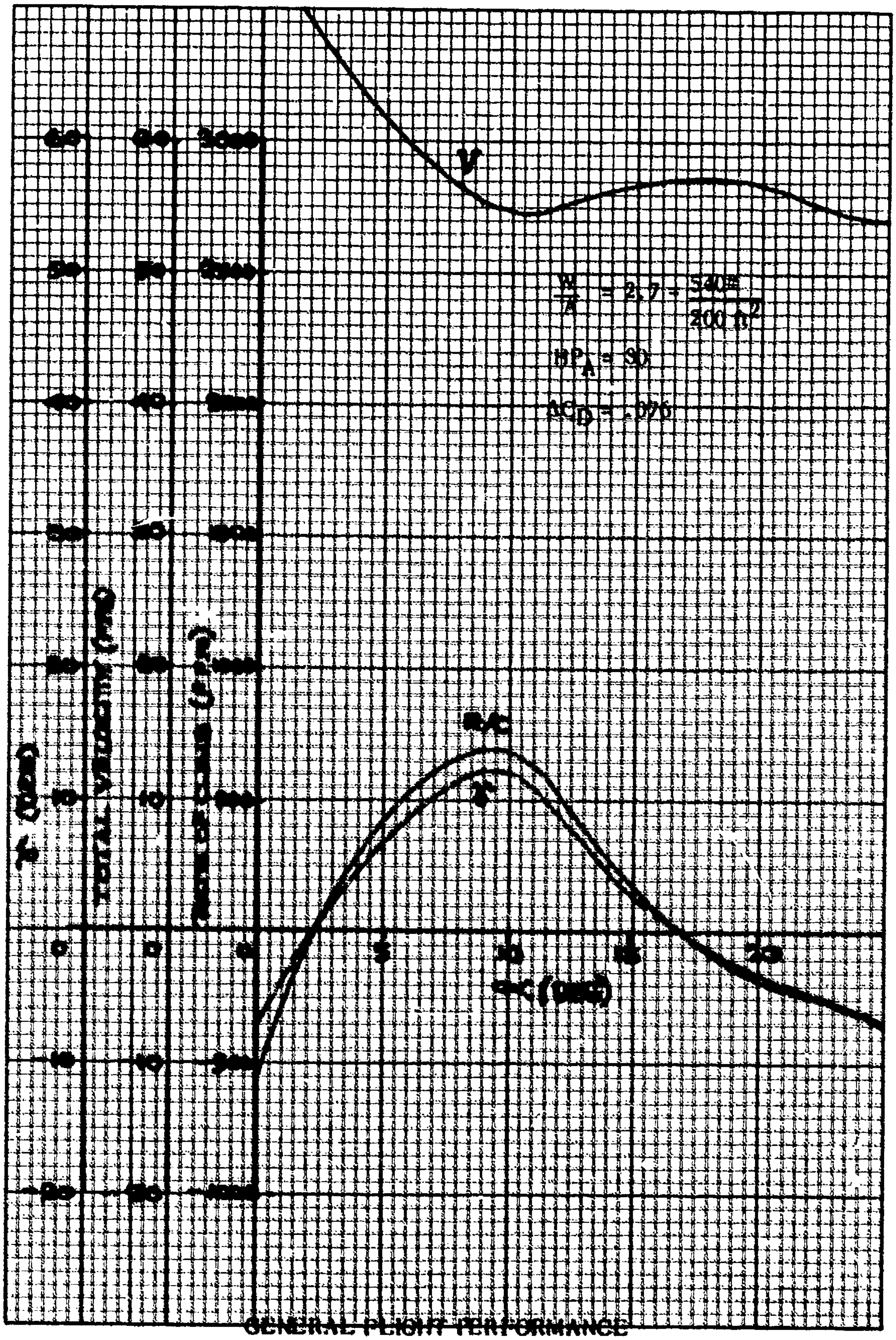

$\frac{W}{A} = \frac{540^{\#}}{200\ ft^2} = 2.7$
$\Delta C_D = .076$
$HP_{A(i)} = 20$
V
GENERAL FLIGHT PERFORMANCE

Figure 19a



GENERAL FLIGHT PERFORMANCE
Figure 19b

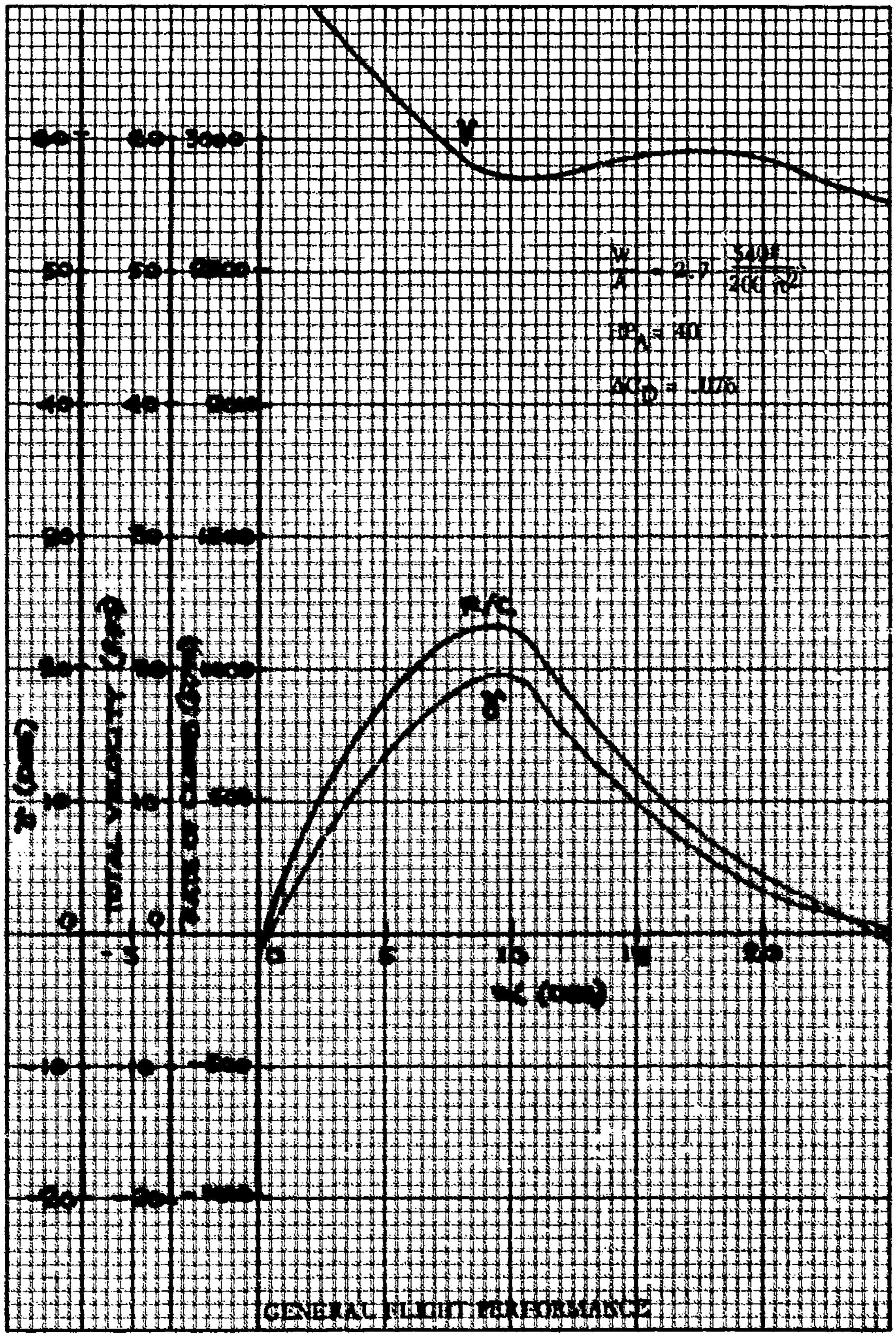


Figure 19c

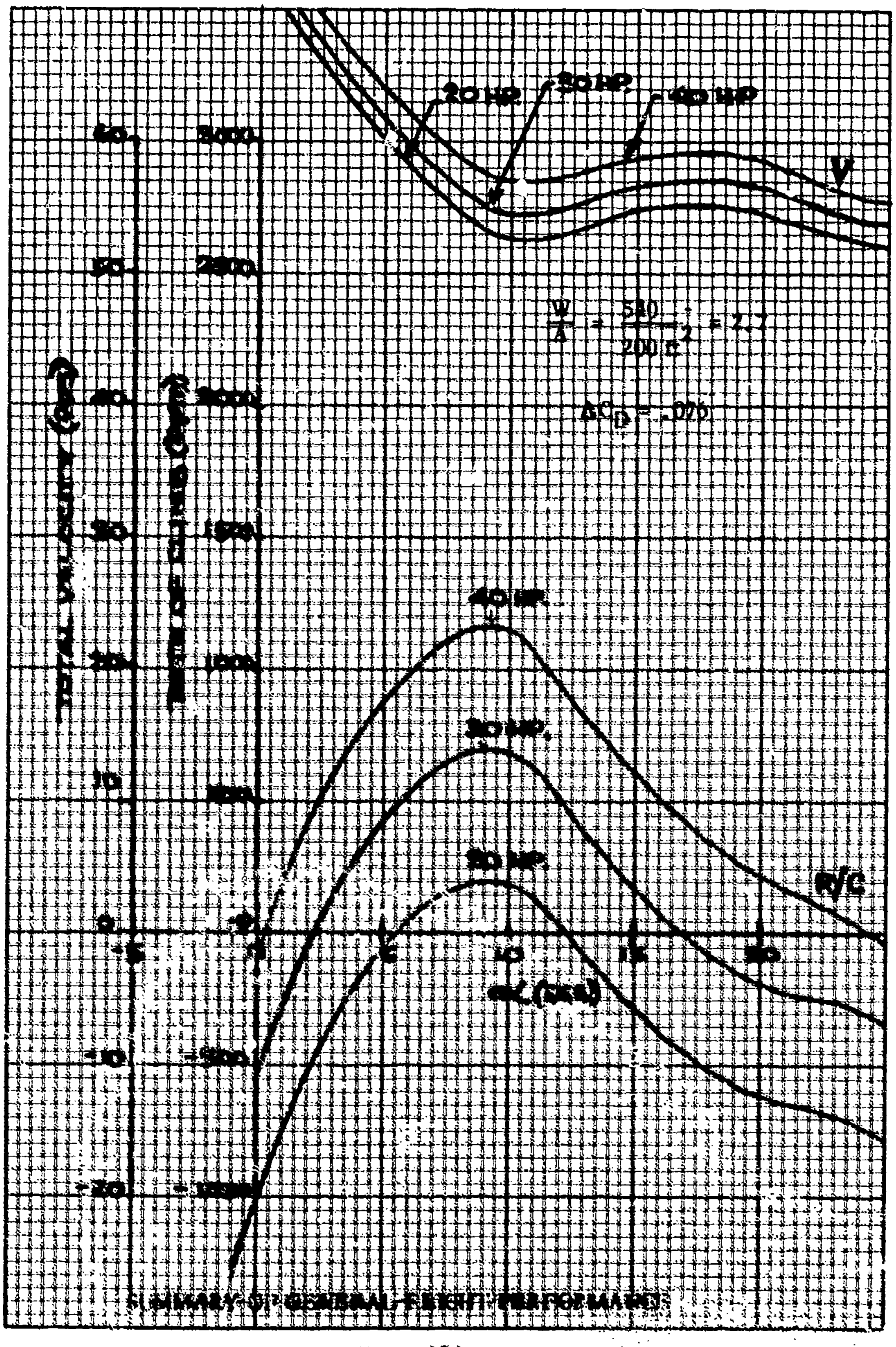
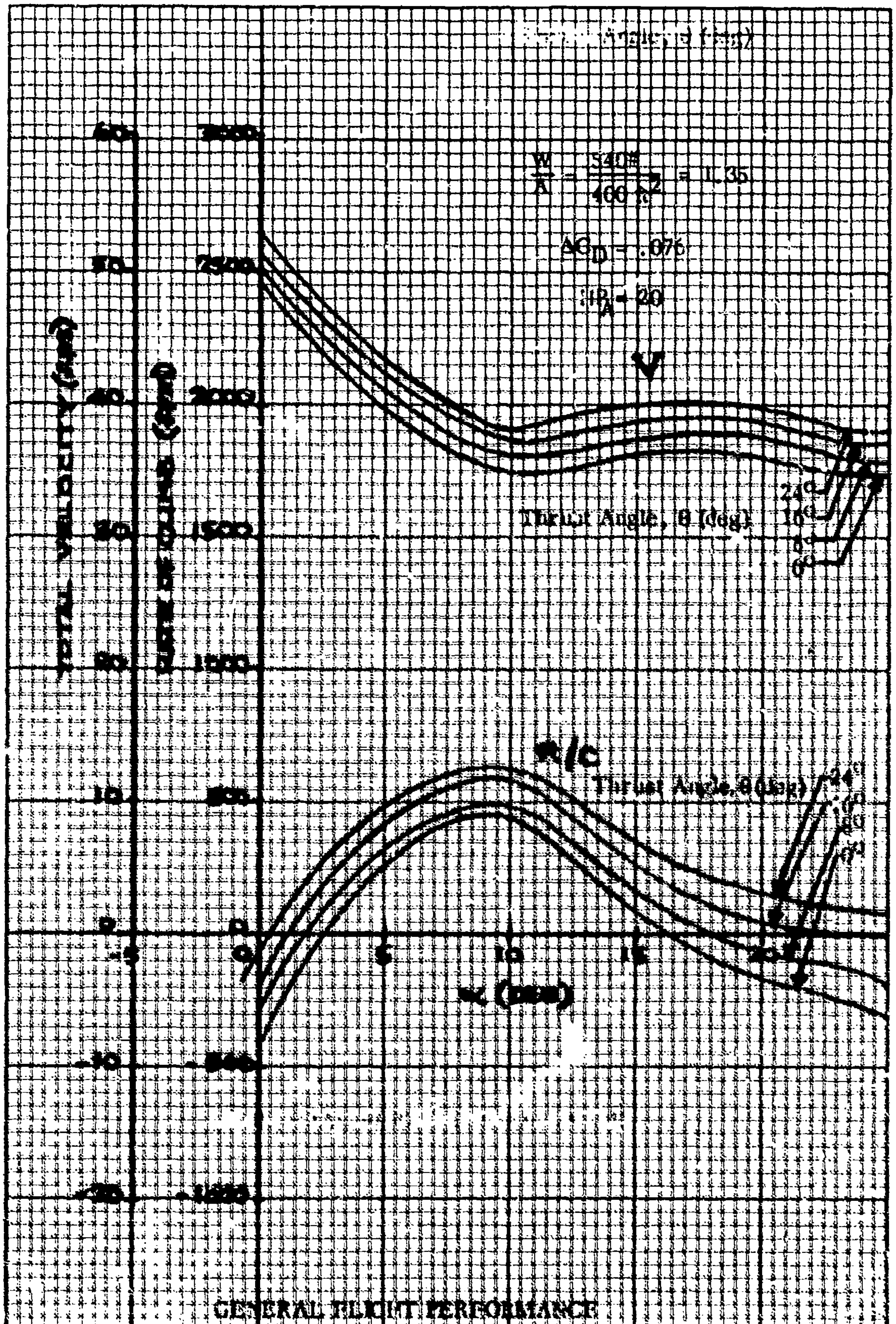

20 HP.
30 HP.
40 HP
V
$\frac{W}{A} = \frac{540}{200\,ft^2} = 2.7$
$\Delta C_D = .070$
40 HP.
30 HP.
20 HP.
R/C

Figure 19d



GENERAL FLIGHT PERFORMANCE
THRUST LINE ANGLE EFFECTS

Figure 20 53

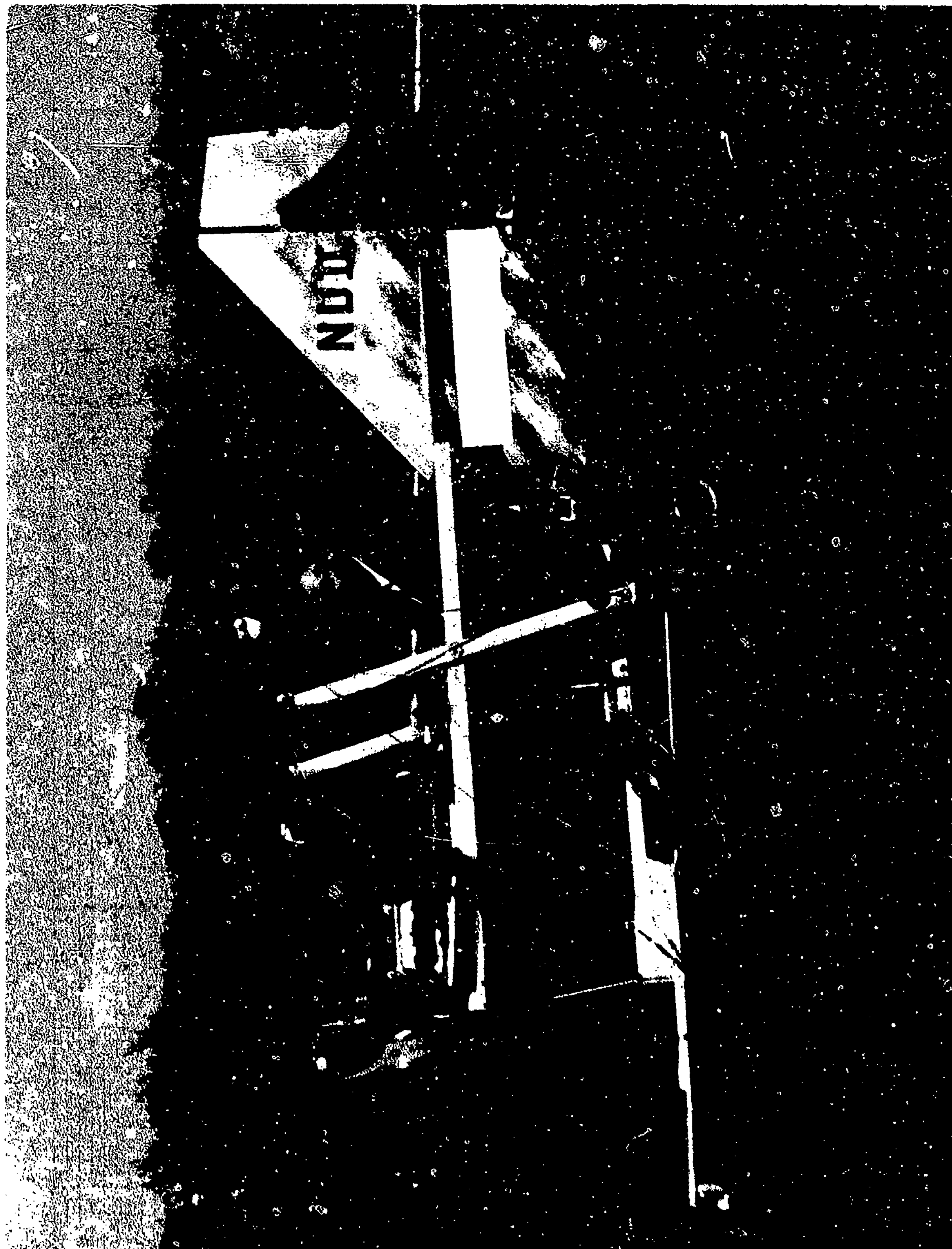
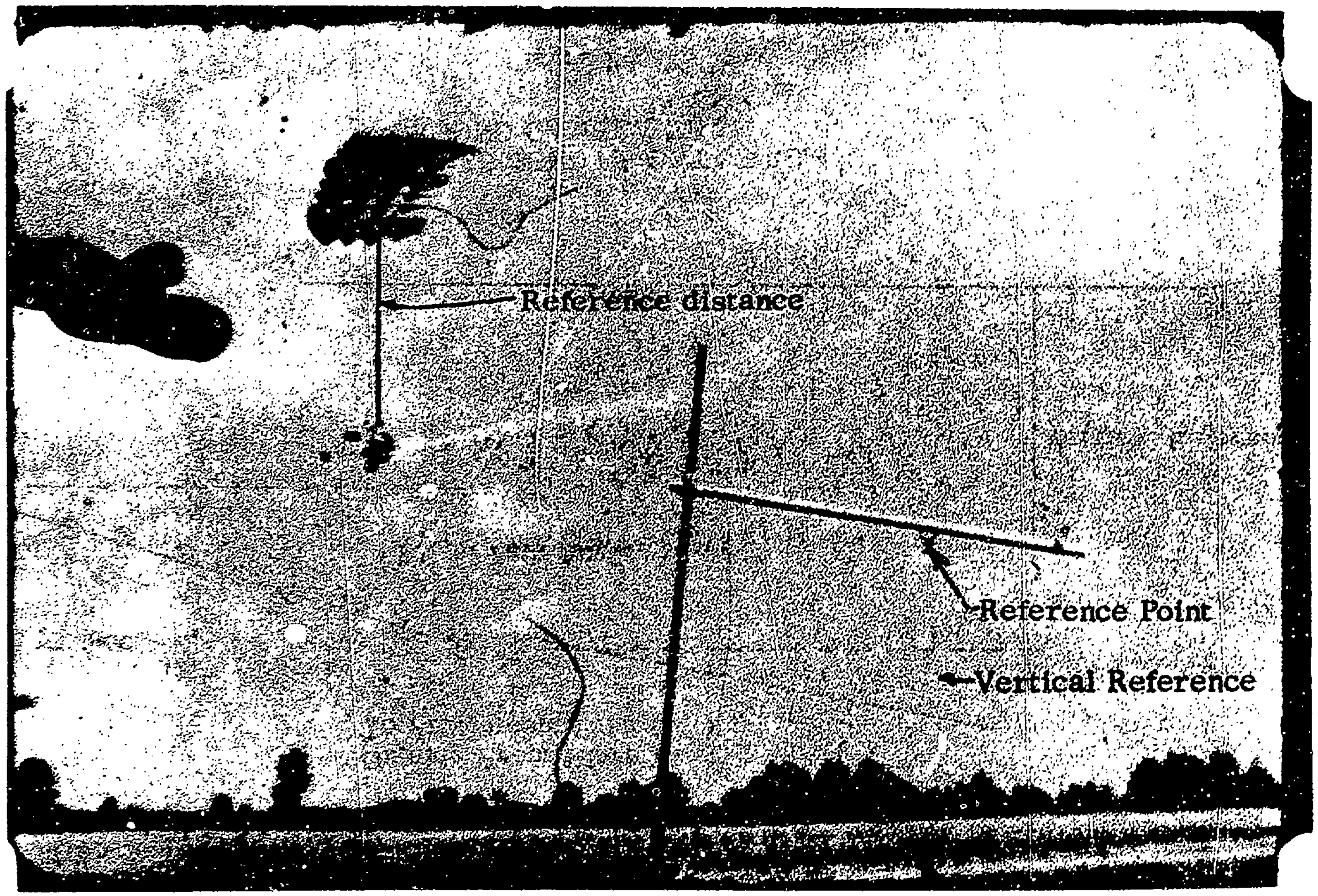
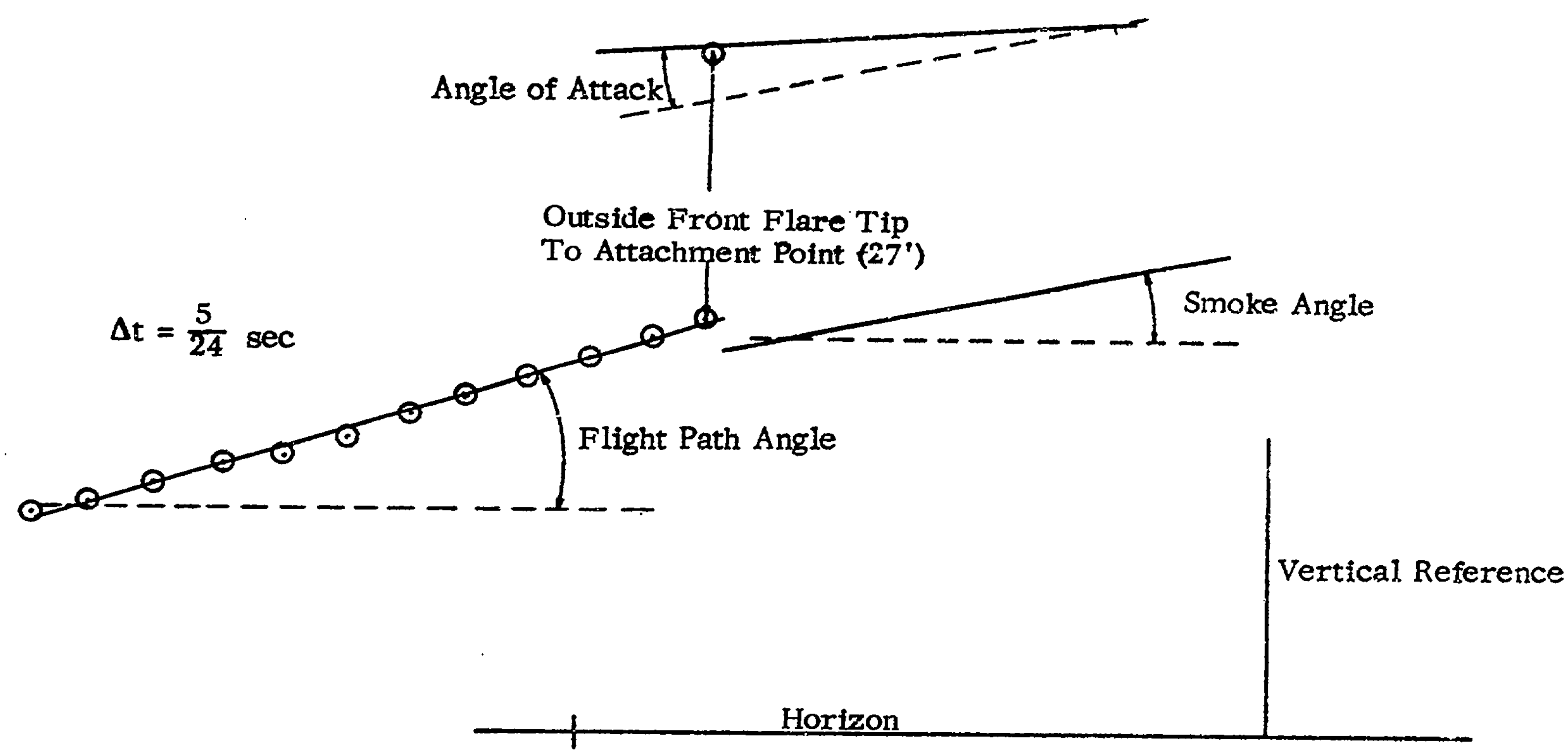


Figure 21. Irish Flyer



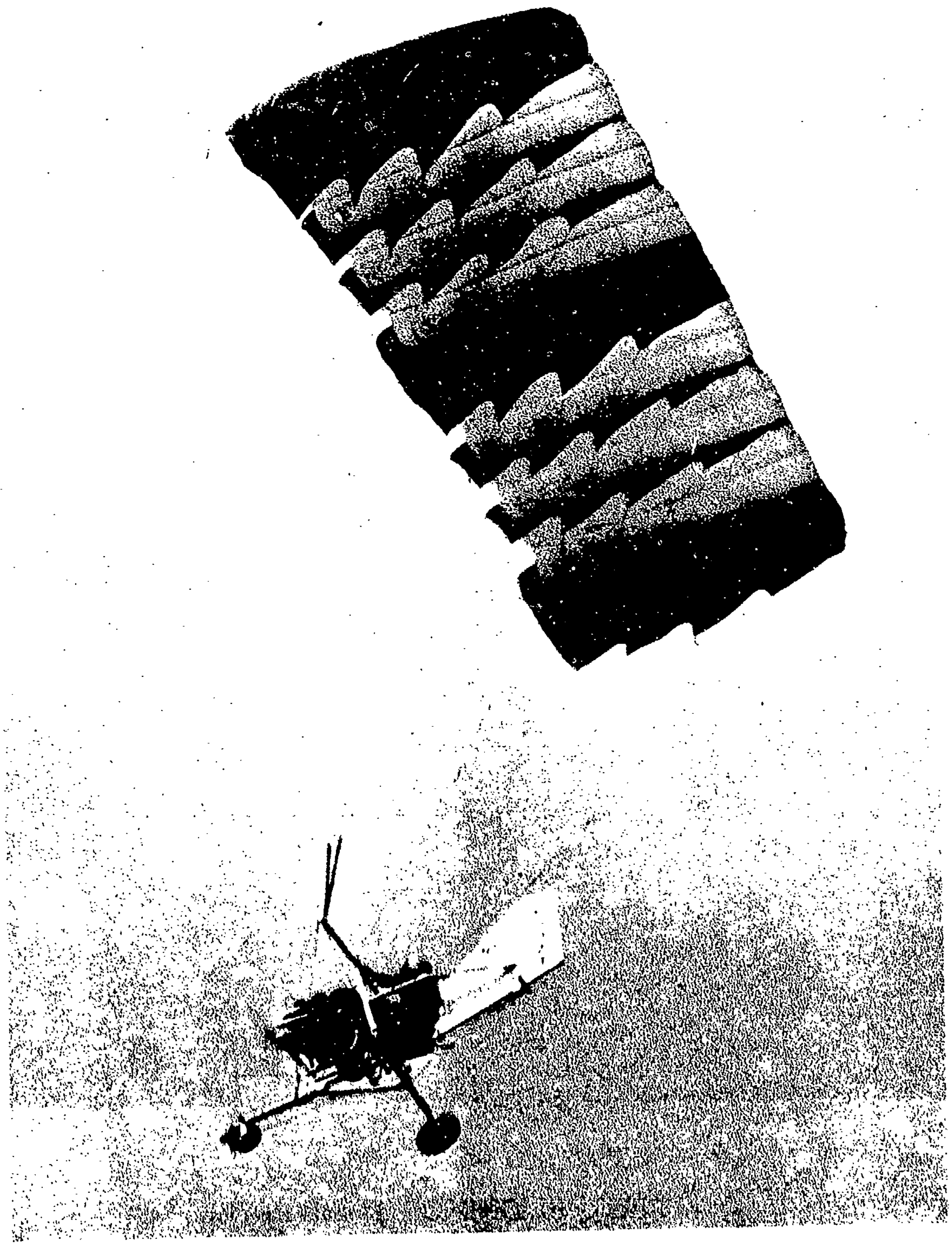
DATA FRAME OF GROUND CAMERA

Figure 22



MEASUREMENT OF DATA FROM
GROUND CAMERA FILM

Figure 23



FIRST FLIGHT OF IRISH FLYER

Figure 24

FIRST FLIGHT OF IRISH FLYER

Figure 24a

APPENDIX A

TABLES OF PERFORMANCE

TABLE I

FLIGHT PARAMETERS FOR W/A = 1.0 ON

400 SQ. FT. PARAFOIL WITH ΔC_D = .038

$\theta = 0^o$ HP_A = 24

α (deg)	C_L	C_D	V (fps)	HP_R	HP_X	R/C (fpm)
-6.0	.077	.168	104.511	165.836		
-5.0	.121	.160	83.371	80.177		
-4.0	.172	.167	69.927	49.378		
-3.0	.226	.171	61.004	33.569		
-2.0	.276	.170	55.202	24.728	-0.728	-60.073
-1.0	.323	.171	51.028	19.647	4.353	359.113
0.0	.377	.174	47.232	15.854	8.146	672.031
1.0	.423	.175	44.590	13.416	10.584	873.152
2.0	.477	.180	41.990	11.524	12.476	1029.274
3.0	.526	.183	39.987	10.118	13.882	1145.295
4.0	.576	.186	38.212	8.974	15.026	1239.646
5.0	.622	.190	36.772	8.169	15.831	1306.049
6.0	.676	.193	35.272	7.324	16.676	1375.777
7.0	.725	.198	34.060	6.765	17.235	1421.891
8.0	.772	.206	33.007	6.405	17.595	1451.553
9.0	.822	.218	31.987	6.170	17.830	1471.010
10.0	.828	.225	31.871	6.299	17.701	1460.366
11.0	.826	.242	31.909	6.799	17.201	1419.074
12.0	.805	.260	32.323	7.593	16.407	1353.617
13.0	.780	.282	32.837	8.634	15.366	1267.693
14.0	.753	.296	33.420	9.554	14.446	1191.758
15.0	.728	.308	33.899	10.458	13.542	1117.192
16.0	.711	.322	34.393	11.328	12.672	1045.430
17.0	.693	.332	34.393	12.138	11.862	978.621
18.0	.688	.350	34.963	12.936	11.064	912.800
19.0	.682	.367	35.117	13.743	10.257	846.165
20.0	.685	.382	35.040	14.211	9.789	807.568
21.0	.695	.404	34.787	14.707	9.293	766.710
22.0	.710	.430	34.418	15.160	8.840	729.334
23.0	.720	.453	34.178	15.639	8.361	689.792
24.0	.722	.477	34.130	16.399	7.601	627.078
25.0	.715	.496	34.297	17.303	6.697	552.478
26.0	.700	.506	34.663	18.223	5.777	476.637
27.0	.678	.522	35.220	19.721	4.279	353.005
28.0	.645	.523	36.110	21.295	2.705	223.200
29.0	.625	.533	36.683	22.752	1.248	102.986
30.0	.605	.545	37.285	24.427	-0.427	-35.226

TABLE II

FLIGHT PARAMETERS FOR W/A = 1.25 ON

400 SQ. FT. PARAFOIL WITH ΔC_D = .038

$\theta = 0^{o}$ $HP_A = 24$

α (deg)	C_L	C_D	V (fps)	HP_R	HP_X	R/C (fpm)
-6.0	.077	.168	116.847	231.76		
-5.0	.121	.160	93.212	112.05		
-4.0	.172	.167	78.181	69.01		
-3.0	.226	.171	68.204	46.91		
-2.0	.276	.170	61.718	34.55		
-1.0	.323	.171	57.051	27.45	-3.458	-228.206
0.0	.377	.174	52.807	22.16	1.843	121.646
1.0	.423	.175	49.853	18.75	5.250	346.507
2.0	.477	.180	46.947	16.11	7.895	521.056
3.0	.526	.183	44.707	14.14	9.860	650.772
4.0	.576	.186	42.722	12.54	11.458	756.259
5.0	.622	.190	41.112	11.42	12.583	830.500
6.0	.676	.193	39.436	10.24	13.765	908.458
7.0	.725	.198	38.080	9.45	14.546	960.016
8.0	.772	.206	36.902	8.95	15.048	993.178
9.0	.822	.218	35.763	8.62	15.378	1014.932
10.0	.828	.225	35.633	8.80	15.197	1003.032
11.0	.826	.242	35.676	9.50	14.498	956.866
12.0	.805	.260	36.138	10.61	13.389	883.683
13.0	.780	.282	36.713	12.07	11.934	787.616
14.0	.753	.296	37.365	13.35	10.647	702.719
15.0	.728	.308	38.001	14.62	9.384	619.352
16.0	.711	.322	38.453	15.85	8.158	539.119
17.0	.693	.332	38.949	16.96	7.037	464.424
18.0	.688	.350	39.090	18.08	5.922	390.834
19.0	.682	.367	39.262	19.21	4.793	316.334
20.0	.685	.382	39.176	19.86	4.139	273.181
21.0	.695	.404	38.893	20.55	3.447	227.501
22.0	.710	.430	38.480	21.19	2.814	185.713
23.0	.720	.453	38.212	21.85	2.144	141.504
24.0	.722	.477	38.159	22.92	1.082	71.387
25.0	.715	.496	38.345	24.13	-0.182	-12.019
26.0	.700	.506	38.754	25.47		
27.0	.678	.522	39.378	27.55		
28.0	.645	.523	40.372	29.76		
29.0	.625	.533	41.013	31.796		
30.0	.605	.545	41.686	34.14		

TABLE III

FLIGHT PARAMETERS FOR W/A = 1.0 ON 400 SQ. FT. PARAFOIL WITH ΔC_D = .076

$\theta = 0^{0}$ $HP_A = 24$

α (deg)	C_L	C_D	V (fps)	HP_R	HP_X	R/C (fpm)
-6	.077	.206	104.511	203.347		
-5	.121	.198	83.371	99.219		
-4	.172	.205	69.927	60.613		
-3	.226	.209	61.004	41.029		
-2	.276	.209	55.202	30.256		
-1	.323	.209	51.028	24.013		
0	.377	.212	47.232	19.317	4.683	386.347
1	.423	.213	44.590	16.330	7.670	632.775
2	.477	.218	41.900	13.957	10.043	828.547
3	.526	.221	39.927	12.219	10.781	889.432
4	.576	.224	38.212	10.807	13.193	1088.422
5	.622	.228	36.772	9.803	14.197	1171.252
6	.676	.231	35.272	8.766	15.224	1255.980
7	.725	.236	34.060	8.063	15.937	1314.802
8	.772	.244	33.007	7.587	16.413	1354.072
9	.822	.256	31.987	7.245	16.755	1382.287
10	.828	.263	31.871	7.362	16.638	1372.635
11	.826	.280	31.909	7.867	16.133	1330.972
12	.805	.298	32.323	8.702	15.298	1262.085
13	.780	.320	32.837	9.797	14.203	1171.747
14	.753	.334	33.420	10.781	10.781	1090.567
15	.728	.346	33.989	11.749	12.251	1010.707
16	.711	.360	34.393	12.665	11.335	935.137
17	.693	.370	34.837	13.527	10.473	864.022
18	.688	.388	34.963	14.340	9.660	796.950
19	.682	.405	35.117	15.166	8.834	728.805
20	.685	.420	35.040	15.625	8.375	690.937
21	.695	.442	34.787	16.090	7.910	652.575
22	.710	.468	34.418	16.499	7.501	618.832
23	.720	.491	34.178	16.951	7.049	581.542
24	.722	.515	34.130	17.705	6.295	519.337
25	.715	.534	34.297	18.629	5.371	443.107
26	.700	.544	34.663	19.591	4.409	363.742
27	.678	.560	35.220	21.157	2.843	234.547
28	.645	.561	36.110	22.842	1.158	95.535
29	.625	.571	36.683	24.374	-0.374	- 30.855
30	.605	.583	37.285	26.130	-2.130	- 175.725

TABLE IV

FLIGHT PARAMETERS FOR W/A = 1.25 ON

400 SQ.FT. PARAFOIL WITH ΔC_D = .076

$\theta = 0^{\circ}$ $HP_A = 24$

α (deg)	C_L	C_D	V (fps)	HP_R	HP_X	R/C (fpm)
-6.0	.077	.206	116.847	284.186		
-5.0	.121	.198	93.212	138.662		
-4.0	.172	.205	78.181	84.710		
-3.0	.226	.209	68.204	57.340		
-2.0	.276	.208	61.781	42.284		
-1.0	.323	.209	57.051	33.559		
0.0	.377	.212	52.807	26.996	-2.996	-197.719
1.0	.423	.213	49.853	22.821	1.179	77.794
2.0	.477	.218	46.947	19.505	4.495	296.657
3.0	.526	.221	44.707	17.076	6.924	456.987
4.0	.576	.224	42.722	15.104	8.896	587.150
5.0	.622	.228	41.112	13.700	10.300	679.800
6.0	.676	.231	39.436	12.251	11.749	775.449
7.0	.725	.236	38.081	11.269	12.731	840.261
8.0	.772	.244	36.902	10.603	13.397	884.192
9.0	.822	.256	35.763	10.125	13.875	915.737
10.0	.828	.263	35.633	10.289	13.711	904.913
11.0	.826	.280	35.676	10.994	13.006	858.390
12.0	.805	.298	36.138	12.162	11.838	781.329
13.0	.780	.320	36.713	13.692	10.308	680.302
14.0	.753	.334	37.365	15.067	8.933	589.581
15.0	.728	.346	38.001	16.419	7.581	500.337
16.0	.711	.360	38.453	17.700	6.300	415.810
17.0	.693	.370	38.949	18.905	5.095	336.280
18.0	.683	.388	39.090	20.041	3.959	261.291
19.0	.682	.405	39.262	21.196	2.804	185.077
20.0	.685	.420	39.176	21.837	2.163	142.785
21.0	.695	.442	38.893	22.486	1.514	38.893
22.0	.710	.468	38.480	23.059	0.942	62.144
23.0	.720	.491	38.212	23.689	0.311	20.499
24.0	.722	.515	38.159	24.744	-0.744	-49.115
25.0	.715	.524	38.345	26.035		
26.0	.700	.544	38.745	27.379		
27.0	.678	.560	39.378	29.568		
28.0	.645	.561	40.372	31.922		
29.0	.625	.571	41.031	34.063		
30.0	.605	.583	41.686	36.518		

TABLE V

FLIGHT PARAMETERS FOR W/A = 2.0 ON

200 SQ. FT. PARAFOIL WITH ΔC_D = .076

$\theta = 0^o$ $HP_A = 24$

α (deg)	C_L	C_D	V (fps)	HP_R	HP_X	R/C (fpm)
-6.0	.077	.206	147.801	287.576		
-5.0	.121	.198	117.905	140.316		
-4.0	.172	.205	98.892	85.720		
-3.0	.226	.209	86.272	58.024		
-2.0	.276	.208	78.067	42.788		
-1.0	.323	.209	72.164	33.960		
0.0	.377	.212	66.796	27.381	-3.318	-273.716
1.0	.423	.213	63.060	23.094	0.906	74.783
2.0	.477	.218	59.383	19.738	4.262	351.626
3.0	.526	.221	56.550	17.280	6.720	554.429
4.0	.576	.224	46.678	10.730	8.716	719.074
5.0	.622	.228	52.003	13.863	10.137	836.267
6.0	.676	.231	49.883	12.397	11.603	957.255
7.0	.725	.236	48.168	11.403	12.597	1039.257
8.0	.772	.244	46.678	10.730	13.270	1094.804
9.0	.822	.256	45.236	10.246	13.754	1134.706
10.0	.828	.263	45.072	10.412	13.588	1121.015
11.0	.826	.280	45.127	11.125	12.875	1062.161
12.0	.805	.298	45.712	12.307	11.693	964.692
13.0	.780	.320	46.438	13.856	10.144	836.902
14.0	.753	.334	47.264	15.247	8.753	722.149
15.0	.728	.346	48.068	16.615	7.385	609.262
16.0	.711	.360	48.640	17.911	6.089	502.344
17.0	.693	.370	49.267	19.130	4.870	401.745
18.0	.688	.388	49.446	20.280	3.720	306.890
19.0	.682	.405	49.663	21.449	2.551	210.487
20.0	.685	.420	49.554	22.097	1.903	156.991
21.0	.695	.442	49.196	22.754	1.246	102.758
22.0	.710	.468	48.647	23.333	0.667	54.987
23.0	.720	.491	48.355	23.972	0.028	2.311
24.0	.722	.515	48.268	25.039	-1.039	-85.754
25.0	.715	.534	48.503	26.345		
26.0	.700	.544	49.020	27.706		
27.0	.678	.560	49.809	29.920		
28.0	.645	.561	51.067	32.303		
29.0	.625	.571	51.878	34.470		
30.0	.605	.583	52.729	36.954		

TABLE VI

FLIGHT PARAMETERS AT 5000 FEET

HORSEPOWER AVAILABLE = 19.768 HP

$\theta = 0^{\circ}$ $\Delta C_D = .038$ $\frac{W}{A} = \frac{400\ \#}{400\ ft^2}$

α (deg)	C_L	C_D	V (fps)	HP_R	HP_X	R/C (fpm)
-6.0	.077	.168	112.590	178.655		
-5.0	.121	.160	89.816	86.374		
-4.0	.172	.167	75.332	53.194		
-3.0	.226	.171	65.719	36.164		
-2.0	.276	.170	59.469	26.640		
-1.0	.323	.171	54.972	21.166	-1.397	-115.286
0.0	.377	.174	50.883	17.080	2.689	221.818
1.0	.423	.175	48.037	14.453	5.315	438.486
2.0	.477	.180	45.236	12.415	7.354	606.675
3.0	.526	.183	43.078	10.900	8.869	731.664
4.0	.576	.186	41.165	9.668	10.101	833.307
5.0	.622	.190	39.614	8.801	10.968	904.843
6.0	.676	.193	37.999	7.890	11.878	979.961
7.0	.725	.198	36.692	7.288	12.480	1029.640
8.0	.772	.206	35.558	6.901	12.868	1061.594
9.0	.822	.218	34.459	6.646	13.122	1082.555
10.0	.828	.225	34.334	6.785	12.983	1071.089
11.0	.826	.242	34.376	7.325	12.444	1026.605
12.0	.805	.260	34.821	8.179	11.589	956.088
13.0	.780	.282	35.375	9.301	10.467	863.522
14.0	.753	.296	36.004	10.293	9.475	781.718
15.0	.728	.308	36.617	11.267	8.502	701.389
16.0	.711	.322	37.052	12.204	7.565	624.090
17.0	.693	.332	37.530	13.076	6.692	552.106
18.0	.688	.350	37.666	13.936	5.833	481.198
19.0	.682	.367	37.831	14.806	4.963	409.412
20.0	.685	.382	37.748	15.310	4.459	367.831
21.0	.695	.404	37.476	15.843	3.925	323.816
22.0	.710	.430	37.078	16.331	3.437	283.551
23.0	.720	.453	36.819	16.848	2.921	240.952
24.0	.722	.477	36.768	17.667	2.102	173.390
25.0	.715	.496	36.948	18.641	1.128	93.024
26.0	.700	.506	37.342	19.631	0.137	11.322
27.0	.678	.522	37.943	21.246	-1.477	-121.867
28.0	.645	.523	38.901	22.941		
29.0	.625	.533	39.519	24.510		
30.0	.605	.545	40.167	26.315		

TABLE VII

FLIGHT PARAMETERS AT 10000 FEET

HORSEPOWER AVAILABLE = 16.169 HP

$\theta = 0^{\circ}$ $\Delta C_D = .038$ $\frac{W}{A} = \frac{400\#}{400 \text{ ft}^2}$

α (deg)	C_L	C_D	V (fps)	HP_R	HP_X	R/C (fpm)
-6.0	.077	.158	121.482	192.765		
-5.0	.121	.160	95.909	93.196		
-4.0	.172	.167	81.282	57.396		
-3.0	.226	.171	57.314	22.837		
-2.0	.276	.170	64.166	28.744		
-1.0	.323	.171	59.314	22.837		
0.0	.377	.174	54.902	18.429	-2.259	-186.398
1.0	.423	.175	51.831	15.595	0.574	47.333
2.0	.477	.180	48.890	13.395	2.774	228,856
3.0	.526	.183	46.480	11.761	4.409	363.717
4.0	.576	.186	44.417	10.431	5.738	473.389
5.0	.622	.190	42.743	9.496	6.674	550.575
6.0	.676	.193	41.000	8.513	7.656	631.625
7.0	.725	.198	39.590	7.863	8.306	685.228
8.0	.772	.206	38.366	7.446	8.724	719.706
9.0	.822	.218	37.181	7.171	8.998	742.323
10.0	.828	.225	37.046	7.321	8.848	729.951
11.0	.826	.242	37.091	7.903	8.266	681.953
12.0	.805	.260	37.572	8.825	7.344	605.867
13.0	.780	.282	38.169	10.036	6.133	505.920
14.0	.753	.296	38.847	11.106	5.063	417.725
15.0	.728	.308	39.509	12.157	4.013	331.051
16.0	.711	.322	39.978	13.168	3.002	247.636
17.0	.693	.332	40.494	14.109	2.060	169.978
18.0	.688	.350	40.641	15.036	1.133	93.469
19.0	.682	.367	40.819	15.975	0.194	16.013
20.0	.685	.382	40.730	16.519	-0.350	-28.352
21.0	.695	.404	40.436	17.095		
22.0	.710	.430	40.006	17.621		
23.0	.720	.453	39.723	18.178		
24.0	.722	.477	39.673	19.062		
25.0	.715	.496	39.566	20.113		
26.0	.700	.506	40.291	21.812		
27.0	.678	.522	40.940	22.924		
28.0	.645	.523	41.974	24.792		
29.0	.625	.533	42.640	26.446		
30.0	.605	.545	43.339	28.394		

TABLE VIII

FLIGHT PARAMETERS FOR $\theta = -20^{o}$

W/A = 1.0 ON 400 SQ. FT. PARAFOIL WITH ΔC_D = .038 (HP_A=24)

α (deg)	C_L	C_D	V (fps)	HP_R	HP_X	R/C (fpm)
-5.0	.121	.160	115.747	214.550		
-4.0	.172	.167	86.956	94.950		
-3.0	.226	.171	71.662	54.417		
-2.0	.276	.170	62.670	36.184		
-1.0	.323	.171	56.791	27.084		
.0	.377	.174	51.780	20.889	1.66	137.14
1.0	.423	.175	48.380	17.137	5.42	446.75
2.0	.477	.180	45.209	14.382	8.17	674.00
3.0	.526	.183	42.787	12.395	10.16	837.93
4.0	.576	.186	40.676	10.825	11.73	967.51
5.0	.622	.190	39.003	9.748	12.80	1056.31
6.0	.676	.193	37.261	8.634	13.92	1148.27
7.0	.725	.198	35.890	7.915	14.64	1207.58
8.0	.772	.206	34.736	7.466	15.09	1244.62
9.0	.822	.218	33.652	7.184	15.37	1267.89
10.0	.828	.225	33.574	7.363	15.19	1253.10
11.0	.826	.242	33.760	8.052	14.50	1196.29
12.0	.805	.260	34.408	9.159	13.39	1104.96
13.0	.780	.282	35.236	10.669	11.88	980.40
14.0	.753	.296	36.102	12.044	10.51	866.92
15.0	.728	.308	36.953	13.439	9.11	751.81
16.0	.711	.322	37.634	14.841	7.11	636.11
17.0	.693	.332	38.339	16.178	6.37	525.78
18.0	.688	.350	38.732	17.585	4.97	409.70
19.0	.682	.367	39.160	19.058	3.49	288.24
20.0	.685	.382	39.248	19.971	2.58	212.90
21.0	.695	.404	39.176	21.006	1.55	127.50
22.0	.710	.430	38.980	22.023	.53	43.57
23.0	.720	.453	38.923	23.098	-.55	-45.20
24.0	.722	.477	39.161	24.772		
25.0	.715	.496	39.667	26.771		

TABLE IX

FLIGHT PARAMETERS FOR $\theta = -10^{\circ}$

W/A = 1.0 ON 400 SQ. FT. PARAFOIL WITH ΔC_D = .038 (HP_A=24)

α (deg)	C_L	C_D	V (fps)	HP_R	HP_X	R/C (fpm)
-6.0	.077	.168	133.229	343.546		
-5.0	.121	.160	95.203	119.386		
-4.0	.172	.167	76.809	65.438		
-3.0	.226	.171	65.530	41.611		
-2.0	.276	.170	58.468	29.381		
-1.0	.323	.171	53.590	22.758	0.88	72.38
.0	.377	.174	49.280	18.007	5.63	464.35
1.0	.423	.175	46.311	15.030	8.60	709.90
2.0	.477	.180	43.461	12.778	10.86	895.76
3.0	.526	.183	41.272	11.125	12.51	1032.07
4.0	.576	.186	39.348	9.799	13.84	1141.52
5.0	.622	.190	37.804	8.876	14.76	1217.62
6.0	.676	.193	36.195	7.914	15.72	1297.03
7.0	.725	.198	34.910	7.285	16.35	1348.93
8.0	.772	.206	33.811	6.886	16.75	1381.86
9.0	.822	.218	32.762	6.629	17.01	1403.03
10.0	.828	.225	32.663	6.780	16.86	1390.58
11.0	.826	.242	32.767	7.362	16.27	1342.55
12.0	.805	.260	33.284	8.290	15.34	1265.96
13.0	.780	.282	33.936	9.531	14.10	1163.64
14.0	.753	.296	34.642	10.641	12.99	1072.01
15.0	.728	.308	35.333	11.748	11.89	980.71
16.0	.711	.322	35.854	12.834	10.80	891.09
17.0	.693	.332	36.409	13.856	9.78	806.81
18.0	.688	.350	36.645	14.894	8.74	721.16
19.0	.682	.367	36.911	15.960	7.68	633.20
20.0	.685	.382	36.901	16.598	7.04	580.57
21.0	.695	.404	36.719	17.296	6.34	522.97
22.0	.710	.430	36.416	17.957	5.68	468.40
23.0	.720	.453	36.247	18.655	4.98	410.84
24.0	.722	.477	36.310	19.746	3.89	320.79
25.0	.715	.496	36.608	21.043	2.59	213.82
26.0	.700	.506	37.107	22.357	1.28	105.40
27.0	.678	.522	37.885	24.545	-.91	-75.10
28.0	.645	.523	39.005	26.838		
29.0	.625	.533	39.797	29.050		
30.0	.605	.545	40.652	31.661		

TABLE X

FLIGHT PARAMETERS FOR $\theta = 10^{o}$

W/A = 1.0 ON 400 SQ.FT. PARAFOIL WITH ΔC_D = .038 (HP_A = 24)

α (deg)	C_L	C_D	V (fps)	HP_R	HP_X	R/C (fpm)
-6.0	.077	.168	88.817	101.783		
-5.0	.121	.160	75.078	58.552		
-4.0	.172	.167	64.615	38.958		
-3.0	.226	.171	57.301	27.821		
-2.0	.276	.170	52.429	21.185	2.45	202.19
-1.0	.323	.171	48.801	17.186	6.45	532.13
.0	.377	.174	45.421	14.099	9.54	786.80
1.0	.423	.175	43.048	12.072	11.56	954.02
2.0	.477	.180	40.660	10.463	13.17	1086.77
3.0	.526	.183	38.814	9.253	14.38	1186.54
4.0	.576	.186	37.169	8.259	15.38	1268.59
5.0	.622	.190	35.820	7.551	16.08	1326.98
6.0	.676	.193	34.417	6.804	16.83	1388.62
7.0	.725	.198	33.268	6.304	17.33	1429.84
8.0	.772	.206	32.257	5.979	17.66	1456.70
9.0	.822	.218	31.264	5.761	17.87	1474.66
10.0	.828	.225	31.134	5.872	17.76	1465.52
11.0	.826	.242	31.116	6.304	17.33	1429.82
12.0	.805	.260	31.440	6.987	16.65	1373.48
13.0	.780	.282	31.838	7.870	15.77	1300.68
14.0	.753	.296	32.319	8.641	14.99	1237.07
15.0	.728	.308	32.789	9.389	14.25	1175.39
16.0	.711	.322	33.097	10.095	13.54	1117.09
17.0	.693	.332	33.453	10.748	12.89	1063.24
18.0	.688	.350	33.494	11.372	12.26	1011.75
19.0	.682	.367	33.561	11.996	11.64	960.25
20.0	.685	.382	33.435	12.347	11.29	931.37
21.0	.695	.404	33.131	12.704	10.93	901.84
22.0	.710	.430	32.715	13.020	10.62	875.83
23.0	.720	.453	32.427	13.356	10.28	848.08
24.0	.722	.477	32.301	13.901	9.73	803.13
25.0	.715	.496	32.374	14.554	9.08	749.30
26.0	.700	.506	32.645	15.222	8.41	694.15
27.0	.678	.522	33.049	16.294	7.34	605.75
28.0	.645	.523	33.777	17.427	6.21	512.24
29.0	.625	.533	34.202	18.440	5.20	428.65
30.0	.605	.545	34.636	19.582	4.05	334.50

TABLE XI

FLIGHT PARAMETERS FOR $\theta = 20^{\circ}$

W/A = 1.0 ON 400 SQ. FT. PARAFOIL WITH ΔC_D = .038 (HP_A=24)

α (deg)	C_L	C_D	V (fps)	HP_R	HP_X	R/C (fpm)
-6.0	.077	.168	78.029	69.018		
-5.0	.121	.160	68.503	44.477		
-4.0	.172	.167	60.110	31.364		
-3.0	.226	.171	54.018	23.308		
-2.0	.276	.170	49.893	18.258	4.30	354.42
-1.0	.323	.171	46.725	15.084	7.47	616.19
.0	.377	.174	43.704	12.560	9.99	824.40
1.0	.423	.175	41.571	10.871	11.68	963.75
2.0	.477	.180	39.374	9.501	13.05	1076.77
3.0	.526	.183	37.673	8.461	14.09	1162.58
4.0	.576	.186	36.147	7.596	14.96	1233.90
5.0	.622	.190	34.884	6.974	15.58	1285.21
6.0	.676	.193	33.572	6.315	16.24	1339.64
7.0	.725	.198	32.484	5.869	16.68	1376.43
8.0	.772	.206	31.512	5.574	16.98	1400.74
9.0	.822	.218	30.547	5.373	17.18	1417.31
10.0	.828	.225	30.403	5.468	17.08	1409.50
11.0	.826	.242	30.333	5.841	16.71	1378.75
12.0	.805	.260	30.576	6.427	16.13	1330.39
13.0	.780	.282	30.869	7.173	15.38	1268.85
14.0	.753	.296	31.259	7.818	14.73	1215.61
15.0	.728	.308	31.641	8.437	14.12	1164.59
16.0	.711	.322	31.868	9.011	13.54	1117.20
17.0	.693	.332	32.147	9.538	13.02	1073.74
18.0	.688	.350	32.117	10.026	12.53	1033.45
19.0	.682	.367	32.113	10.510	12.04	993.57
20.0	.685	.382	31.948	10.771	11.78	972.00
21.0	.695	.404	31.605	11.028	11.52	950.80
22.0	.710	.430	31.155	11.245	11.31	932.97
23.0	.720	.453	30.830	11.479	11.07	913.63
24.0	.722	.477	30.645	11.871	10.68	881.33
25.0	.715	.496	30.646	12.345	10.21	842.18
26.0	.700	.506	30.843	12.838	9.72	801.56
27.0	.678	.522	31.129	13.615	8.94	737.39
28.0	.645	.523	31.731	14.449	8.10	668.65
29.0	.625	.533	32.046	15.168	7.39	609.28
30.0	.605	.545	32.357	15.965	6.59	543.59

TABLE XII

FLIGHT PARAMETERS FOR $\theta = 30^{\circ}$

W/A = 1.0 ON 400 SQ.FT. PARAFOIL WITH ΔC_D = .038 (HP_A=24)

α (deg)	C_L	C_D	V (fps)	HP_R	HP_X	R/C (fpm)
-6.0	.077	.168	69.529	48.831		
-5.0	.121	.160	62.785	34.243		
-4.0	.172	.167	55.979	25.331		
-3.0	.226	.171	50.894	19.493	1.29	106.77
-2.0	.276	.170	47.413	15.668	5.12	422.22
-1.0	.323	.171	44.659	13.170	7.62	628.30
.0	.377	.174	41.971	11.125	9.67	797.03
1.0	.423	.175	40.062	9.730	11.05	912.04
2.0	.477	.180	38.050	8.575	12.21	1007.36
3.0	.526	.183	36.490	7.689	13.10	1080.45
4.0	.576	.186	35.082	6.945	13.84	1141.85
5.0	.622	.190	33.904	6.403	14.38	1186.52
6.0	.675	.193	32.682	5.826	14.96	1234.12
7.0	.725	.198	31.656	5.431	15.35	1266.68
8.0	.772	.206	30.725	5.167	15.62	1288.50
9.0	.822	.218	29.783	4.983	15.80	1303.69
10.0	.828	.225	29.632	5.062	15.72	1297.14
11.0	.826	.242	29.511	5.379	15.41	1271.03
12.0	.805	.260	29.675	5.875	14.91	1230.06
13.0	.780	.282	29.868	6.497	14.29	1178.73
14.0	.753	.296	30.172	7.031	13.75	1134.76
15.0	.728	.308	30.472	7.536	13.25	1093.09
16.0	.711	.322	30.623	7.996	12.79	1055.12
17.0	.693	.332	30.834	8.416	12.37	1020.49
18.0	.688	.350	30.740	8.792	11.99	989.49
19.0	.682	.367	30.675	9.160	11.63	959.13
20.0	.685	.382	30.477	9.351	11.43	943.38
21.0	.695	.404	30.102	9.529	11.26	928.70
22.0	.710	.430	29.626	9.669	11.12	917.11
23.0	.720	.453	29.273	9.826	10.96	904.16
24.0	.722	.477	29.039	10.101	10.68	881.49
25.0	.715	.496	28.982	10.441	10.34	853.45
26.0	.700	.506	29.116	10.800	9.99	823.80
27.0	.687	.522	29.306	11.361	9.43	777.60
28.0	.645	.523	29.803	11.972	8.81	727.17
29.0	.625	.533	30.029	12.481	8.31	685.18
30.0	.605	.545	30.242	13.035	7.75	639.47

TABLE XIII

FLIGHT PARAMETERS FOR $\theta = 40^{\circ}$

W/A = 1.0 ON 400 SQ.FT. PARAFOIL WITH ΔC_D = .038 (HP_A= 24)

α (deg)	C_L	C_D	V (fps)	HP_R	HP_X	R/C (fpm)
-6.0	.077	.168	62.123	34.829		
-5.0	.121	.160	57.405	26.173		
-4.0	.172	.167	51.912	20.202		
-3.0	.226	.171	47.713	16.061	2.33	191.94
-2.0	.276	.170	44.823	13.239	5.15	424.73
-1.0	.323	.171	42.463	11.321	7.07	582.87
.0	.377	.174	40.103	9.704	8.68	716.28
1.0	.423	.175	38.419	8.581	9.80	808.87
2.0	.477	.180	36.596	7.629	10.76	887.47
3.0	.526	.183	35.181	6.891	11.49	948.34
4.0	.576	.186	33.896	6.264	12.12	1000.06
5.0	.622	.190	32.808	5.802	12.58	1038.15
6.0	.675	.193	31.682	5.307	13.08	1078.94
7.0	.725	.198	30.722	4.965	13.42	1107.21
8.0	.772	.206	29.836	4.731	13.65	1126.48
9.0	.822	.218	28.930	4.565	13.82	1140.22
10.0	.828	.225	28.761	4.629	13.76	1134.91
11.0	.826	.242	28.589	4.890	13.50	1113.38
12.0	.805	.260	28.671	5.299	13.09	1079.63
13.0	.780	.282	28.764	5.803	12.58	1038.06
14.0	.753	.296	28.982	6.231	12.15	1002.77
15.0	.728	.308	29.200	6.631	11.75	969.74
16.0	.711	.322	29.278	6.988	11.40	940.28
17.0	.693	.332	29.423	7.313	11.07	913.54
18.0	.688	.350	29.271	7.590	10.80	890.62
19.0	.682	.367	29.149	7.860	10.53	868.42
20.0	.685	.382	28.922	7.992	10.39	857.55
21.0	.695	.404	28.521	8.105	10.88	848.17
22.0	.710	.430	28.027	8.186	10.20	841.52
23.0	.720	.453	27.651	8.282	10.10	833.63
24.0	.722	.477	27.377	8.464	9.92	818.62
25.0	.715	.496	27.269	8.697	9.69	799.40
26.0	.700	.506	27.349	8.950	9.44	778.48
27.0	.678	.522	27.454	9.340	9.05	746.34
28.0	.645	.523	27.858	9.777	8.61	710.26
29.0	.625	.533	28.008	10.127	8.26	681.45
30.0	.605	.545	28.139	10.500	7.89	650.64

TABLE XIV

ASCENDING FLIGHT

$\frac{W}{A} = \frac{400\#}{400\ ft^2} = 1$ $\theta = 0^{o}$ $L/D = 2.95$

$\alpha_T = 11^{o}$ $C_L = .826$ $C_{\bar{D}} = .280$ $\Delta C_D = .076$

η	$-\gamma$ (deg)	V (fps)	u (fps)	-w (fps)	T* (lbs)	HP**	R/C (fpm)
0.0	-18.725	31.053	29.410	-9.969	0.0	0.0	-598.162
0.5	- 9.106	31.274	30.880	-4.950	67.797	3.806	-296.971
1.0	0.0	31.909	31.909	0.0	135.593	7.867	0.0
1.5	8.226	32.891	32.553	4.706	203.390	12.038	282.372
2.0	15.410	34.133	32.906	9.070	271.186	16.225	544.198
2.5	21.553	35.553	33.067	13.061	338.983	20.380	783.680
3.0	26.755	37.086	33.115	16.696	406.780	24.492	1001.730
3.5	31.147	38.682	33.105	20.008	474.576	28.565	1200.496
4.0	34.864	40.307	33.072	23.042	542.373	32.613	1382.50
4.5	38.026	41.939	33.036	25.836	610.169	36.650	
5.0	40.732	43.563	33.010	28.426	677.966	40.690	

$* \ T = \frac{T_A}{\cos(\gamma - \theta}$

** This is the HP which will yield the R/C as indicated.

TABLE XV

ASCENDING FLIGHT

$\frac{W}{A} = \frac{540}{400} = 1.35 \quad \theta = 0^o \quad L/D = 2.95$

$\alpha_T = 11^o \quad C_L = .826 \quad C_D = 2.80 \quad \Delta C_D = .076$

η	$-\gamma$ (deg)	V (fps)	u (fps)	-w (fps)	T* (lbs)	HP*	R/C (fpm)
0.0	-18.725	36.081	34.171	-11.583	0.0	0.0	-695.001
0.5	- 9.106	36.337	35.879	- 5.751	91.525	5.971	-345.049
1.0	- 0.0	37.075	37.075	0.0	183.051	12.339	0.0
1.5	8.226	38.216	37.823	5.468	274.576	18.882	328.086
2.0	15.410	39.659	38.233	10.538	366.102	25.449	682.301
2.5	21.553	41.309	38.420	15.176	457.627	31.968	910.561
3.0	26.155	43.090	38.476	19.398	549.153	38.417	1163.906
3.5	31.147	44.944	38.465	23.947	640.678	44.806	1394.850
4.0	34.864	46.833	38.426	26.772	732.203	51.156	1606.321
4.5	38.026	48.729	38.385	30.019	823.729	57.488	1801.121
5.0	40.732	50.615	38.354	33.029	915.254	63.824	1981.714

*See footnotes Table XIV.

TABLE XVI

ASCENDING FLIGHT

$\frac{W}{A} = \frac{540\#}{200\ ft^2} = 2.7$ $\theta = 0^{o}$ $L/D = 2.95$

$\alpha_T = 11^{o}$ $C_L = .826$ $C_D = .280$ $\Delta C_D = .076$

η	$-\gamma$ (deg)	V (fps)	u (fps)	$-w$ (fps)	T^* (lbs)	HP^*	R/C (fpm)
0.0	-18.725	51.026	48.325	-16.381	0.0	0.0	-982.880
0.5	- 9.106	51.389	50.741	- 8.133	91.525	8.444	-487.973
1.0	0.0	52.433	52.433	0.0	183,051	17.451	0.0
1.5	8.226	54.045	53.489	7.733	274.576	26.703	463.984
2.0	15.410	56.086	54.069	14.903	366.102	35.991	894.209
2.5	21.553	58.420	54.334	21.462	457.627	45.209	1287.728
3.0	26.755	60.938	54.414	27.434	549.702	54.333	1646.011
3.5	31.147	63.561	54.397	32.877	640.678	63.366	1972.616
4.0	34.864	66.232	54.343	37.861	732.203	72.345	2271.682
4.5	38.026	68.913	54.284	42.453	823.729	81.301	2547.170
5.0	40.732	71.581	54.240	46.709	915.254	90.261	2802.566

*See footnotes Table XIV.

TABLE XVII

CONSTANT HORSEPOWER ASCENDING FLIGHT

$HP_A = 20$ $\theta = 0, \Delta C_D = .076$ $\frac{W}{A} = \frac{540}{400} = 1.35$

α (deg)	C_L	C_D	V (fps)	-w (fps)	u (fps)	$-\gamma$ (deg)	T* (lbs)	R/C (fpm)
-6	.077	.206	74.62	-55.00	50.43	-47.47	218.45	-3300.17
-5	.121	.198	72.14	-45.05	56.34	-38.64	195.72	-2703.30
-4	.172	.205	67.08	-34.08	57.78	-30.53	190.90	-2044.85
-3	.226	.209	62.56	-24.65	57.50	-23.20	191.83	-1479.06
-2	.276	.208	59.10	-17.40	56.48	-17.12	195.26	-1044.08
-1	.323	.209	56.12	-12.11	54.79	-12.46	201.24	- 726.85
	.377	.212	53.06	- 7.49	52.53	- 8.11	209.84	- 449.43
1	.423	.213	50.84	- 4.20	50.67	- 4.73	218.05	- 251.99
2	.477	.218	48.45	- 1.42	48.45	- 1.67	227.91	- 85.21
3	.526	.221	46.61	.72	46.61	.89	236.81	43.43
4	.576	.224	44.93	2.53	44.85	3.23	245.94	152.21
5	.622	.228	43.52	3.86	43.35	5.09	254.38	231.86
6	.676	.231	42.07	5.31	41.73	7.25	264.77	318.73
7	.725	.236	40.85	6.27	40.37	8.83	273.52	376.41
8	.772	.244	39.77	6.91	39.16	10.01	281.81	415.16
9	.822	.256	38.69	7.36	37.98	10.97	290.42	441.99
10	.828	.263	38.53	7.17	37.86	10.73	291.43	430.63
11	.826	.280	38.45	6.41	37.91	9.60	291.15	385.06
12	.805	.298	38.71	5.22	38.36	7.75	287.98	313.56
13	.780	.320	39.00	3.69	38.83	5.43	283.95	221.51
14	.753	.334	39.40	2.42	39.33	3.52	280.39	145.20
15	.728	.346	39.79	1.22	39.77	1.76	277.35	73.54
16	.711	.360	39.99	.14	39.99	.20	275.85	8.59
17	.693	.370	40.26	- .83	40.25	- 1.18	274.14	- 49.84
18	.688	.388	40.16	- 1.70	40.12	- 2.43	275.01	- 102.26
19	.682	.405	40.09	- 2.54	40.01	- 3.64	275.83	- 152.96
20	.685	.420	39.86	- 2.99	39.74	- 4.30	277.65	- 179.46
21	.695	.442	39.42	- 3.41	39.27	- 4.96	281.00	- 204.64
22	.710	.468	38.86	- 3.75	38.67	- 5.54	285.30	- 225.14
23	.720	.491	38.43	- 4.12	38.21	- 6.15	288.75	- 247.35
24	.722	.515	38.14	- 4.73	37.84	- 7.13	291.54	- 284.28
25	.715	.534	38.04	- 5.46	37.65	- 8.26	293.06	- 328.16
26	.700	.544	38.17	- 6.21	37.66	- 9.36	292.98	- 372.86
27	.678	.560	38.33	- 7.34	37.62	-11.04	293.30	- 440.66
28	.645	.561	38.84	- 8.52	37.89	-12.68	291.18	- 511.65
29	.625	.571	39.03	- 9.48	37.86	-14.06	291.40	- 569.11
30	.605	.583	39.19	-10.49	37.76	-15.52	292.17	- 629.42

*See footnotes Table XIV.

APPENDIX B

IRISH FLYERS

IRISH FLYERS

Three Irish Flyers have been constructed and test flown in order to explore the basic feasibility of powered Parafoil flight.

Irish Flyer I

Irish Flyer I was configured by modifying a standard Benson Gyrocopter (Figure B-1 and B-2). The rotor was removed and replaced by a 6 foot cross member to which the Parafoil was attached. Also, the propeller was shrouded in order to avoid entanglement with the Parafoil lines. Irish Flyer I was tested in the summer of 1968 by towing it aloft and releasing it for extended powered glides. Complete flight stability was obtained in all six flights; however, only limited periods of straight and level flight were demonstrated.

Irish Flyer II

Irish Flyer II was constructed in 1971 (Figure B-3).* The results of the various test flights are given and discussed in the body of this report. Figure B-4 shows the suborbital paths for each of the five flights.

Irish Flyer III

Irish Flyer III was also constructed and flight tested in 1971 (Figures B-5 and B-6).** This vehicle utilizes a North American Rockwell JLO-LB-600 engine with a 46 inch diameter propeller. The total vehicle weight with pilot is 400 pounds.*** This pusher concept incorporates a provision for pilot ejection seat recovery and powered flight (Figure B-7). The trim, control and flight stability were first checked out by direct tow tests (Figure B-8) and by numerous ascending and gliding flights. Powered flights with the Irish Flyer III were then carried out (Figures B-9 and B-10). The various flights are discussed in the following paragraphs and the suborbital paths are shown in Figures (B-11 and B-12):

On Saturday December 11, 1971 three powered Parafoil flights in the Irish Flyer III were carried out at the Gosh Airport, Goshen, Indiana:

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*Non-powered test pilot Michael Higgins. Powered test pilot Lowell Farrand. Design and construction Wayne Ison.

**Non-powered and powered test pilot Ed Tavares. Design and construction Wayne Ison.

***The FAA/SAC of 21 January 1972 assigns N-302ND to Nicolaides Irish Flyer. The engine is rated at 20 HP at 3500 RPM.

First Flight

A tow type take-off was utilized. The Irish Flyer III left the ground at an airspeed of 24 mph and was towed to an altitude of approximately 500 feet. The engine was running at 2000 RPM during tow take-off. After the tow line was released the power was increased. The flyer did not climb or maintain level flight. After an extended powered glide of 1/2 mile, the pilot switched the engine off and glided to a landing in a plowed field.

Second Flight

The engine was adjusted and the gas tank was moved to provide better gas flow. Tow take-off and climb were normal. After tow line release, power was added and level flight was achieved. The attitude of the vehicle was slightly nose up. Short periods of climb were attempted successfully. Only 3300 RPM was needed for level flight. Full power is 3500 RPM. The propeller torque caused the craft to turn to the right. After traveling about 3/4 mile, the pilot reduced power and began his descent. Power was switched off at an altitude of 30 feet. Because of the torque and turn problem, Irish Flyer III landed in a corn field instead of on the grass runway.

Third Flight

After being towed to altitude and released, Irish Flyer III maintained level flight. A slow and wide 360° turn to the right was initiated. Short periods of climb were achieved. After a full circle of the airport, the pilot reduced power and established his approach glide. Power was switched off at an altitude of 30 feet. The pilot was able to land at a spot of his own choosing, and the landing was normal. The total distance of the flight was approximately 2 miles.

On Sunday, December 12, 1971 the powered Parafoil flights of the Irish Flyer III were continued at the Goshen Airport. The control lines were extended eight inches for this flight in order to remove a flap deflection which was observed in the tests of the previous day.

Fourth Flight

A tow type take-off was again utilized. Irish Flyer III lifted off the ground at an airspeed of approximately 34 mph. The

engine was running at 2000 RPM while the craft was towed to an altitude of 500 feet. After the tow line was released, full power was added and a slow climb began. However, the increased engine torque again caused a turn to the right, and the vehicle leveled off. Straight flight could not be achieved even by using full left control deflection. Partial climbing was achieved by pulling back on the trim control stick. Full power was required for level flight when the stick was in the normal (neutral) position. The pilot made six complete circuits of the airport. He then reduced power and established his glide. At an altitude of approximately 100 feet, the pilot again applied full power and began to climb in order to avoid a runway marker. He then landed in the normal manner. The total distance of the flight was approximately 12 miles.

Discussion of Irish Flyer Flights

Irish Flyer I

The construction and flights of Irish Flyer I were carried out at the invitation of Life Magazine and were recorded therein.[21] A review of the flight films revealed that the control lines were approximately 3 feet too short. This error caused an excessively large Parafoil trim angle and thus excess drag. The engine power available was simply unable to overcome this large drag for any reasonable time. Because of the damage to the craft on the last landing no further tests were carried out.

Irish Flyer II

The flights of Irish Flyer II are discussed in the main body of the report. In summary, completely stable flight was obtained. Complete control was also demonstrated; right turn, left turn, and full flare landing. Irish Flyer II was able to fly level and it demonstrated a limited ability to climb. The available horsepower of approximately 12 HP was just about equal to the drag.

Irish Flyer II has been modified for the installation of a new engine having at least twice the previous power. It should be available for flight demonstration on 17 March 1972.

Irish Flyer III

Irish Flyer III was designed as a super-light pusher configuration in order to demonstrate basic capability in four areas; 1. powered pilot recovery, 2. powered stand-off guided delivery of ordnance or cargo, 3. special military applications, and 4. sport flying.

Level flight Irish Flyer III performance curves for horsepower vs velocity were computed and are given in Figure B-13 and B-14. They are based on the NASA flap data (Figure B-15) and the Notre Dame flare data. These curves are useful in attempting to understand the Saturday flights (1-3) and the Sunday flight (4).[14]

Considering first Figure B-13 we note that for the Saturday take-off velocity of approximately 24 MPH, the Parafoil has a trim angle of 8° with a flap deflection of 1/3. The horsepower required for flight is 9.5 HP. However when the tow line was released and power was applied, the engine propeller torque produced a right turn which required a full left control deflection. This control deflection was seen to produce a measured trim of

the Parafoil of 11^{o} and a 2/3 flap deflection. The resulting flight point is shown in Figure B-13 which yields a flight velocity of 21 MPH at a required horsepower of 9.0 HP.

On Sunday the original 1/3 flap deflection was removed. As a result the take-off speed increased to 34 MPH which indicates a Parafoil trim of 0^{o} and a required horsepower of 19.5 HP. After release from tow and the application of power, it was again necessary to apply left control. As a result the new trim of the Parafoil was measured to be 4^{o}. Thus, the curves now yield the horsepower required for flight of 10.5 HP and a flight velocity of 25.5 MPH.

Now considering Figure B-14, the Saturday take-off speed of 24 MPH yields a trim of 6^{o} on the 1/2 flap deflection curve and, thus, a required horsepower of 9.5 HP. After tow release and the application of power, a left control deflection was required which produced an observed flight trim of 11^{o}. Using the full flap curve we read a level flight velocity of 18 MPH at a required horsepower of 7.0.

For the measured Sunday take-off speed of near 34 MPH, the performance curves indicate a trim of 0^{o} and a required horsepower of 20 HP. After tow release and the application of power, a left deflection was required. The observed flight trim of 4^{o} using the 1/2 flap curve, yields a flight velocity of 26 MPH and a required horsepower of 10.5 HP.

Both the Notre Dame and the NASA data yield similar flight performance estimates. The Saturday flights are seen to require less horsepower than the Sunday flight. However, the Saturday flight trim of 11^{o} at large flap deflection is not desirable. Also, the Sunday take-off trim of 0^{o} and the Sunday flight trim of 4^{o} is equally undesirable.

The preceeding analysis suggests that future flights would be enhanced if a trim of approximately 6^{o} on the zero flap curve was used. Thus, the take-off speed should be reduced to near 26 MPH at a required horsepower of 9.5 HP. The left control may be reduced by yawing the engine and by providing a method of introducing slight wing warp trim. It is, therefore, estimated that the improved flight condition may be represented by a trim of 8^{o} at zero flap deflection which yields a required horsepower near 9 HP (NASA) or 7.5 HP (ND). This flight condition should yield a flight velocity of 25 MPH.

Also, by using the trim stick control and the magic flare controls, the flight speed may be changed in flight to below 20 MPH or increased to over 35 MPH depending on the engine output. The engine output has been increased since the last flight by utilizing ram air carburetor intakes and by removing the cooling blower from the engine. The latter change has also reduced the engine weight by 30 pounds. Additional engine time should also improve its output.

Thus Irish Flyer III, which flew 12 miles on its last flight, should be able to yield improved performance in the next test series.

Summary

The Irish Flyers have demonstrated stable flight, controllability, and flare landing capability.

Irish Flyer II now has a new engine and is ready for flight testing. Irish Flyer III has been optimized in trim and power and, thus, is also ready for flight testing.*

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*On 27 April 1972 Irish Flyer III was officially flown at Notre Dame for the U.S. Air Force which was represented by Col. Charles Scolatti, Director of Air Force Flight Dynamics Laboratory; Lt.Col.Ernest J.Cross,Jr., Chief, Prototype Division; Mr. Leo Hildebrandt, Chief, Vehicle Equipment Division; Mr. Harley Walker, Aerospace Engineer and Lt. Col. William L. Gaiser, Chief of the Optical Weapons Delivery Systems Branch, Air Force Armament Laboratory.

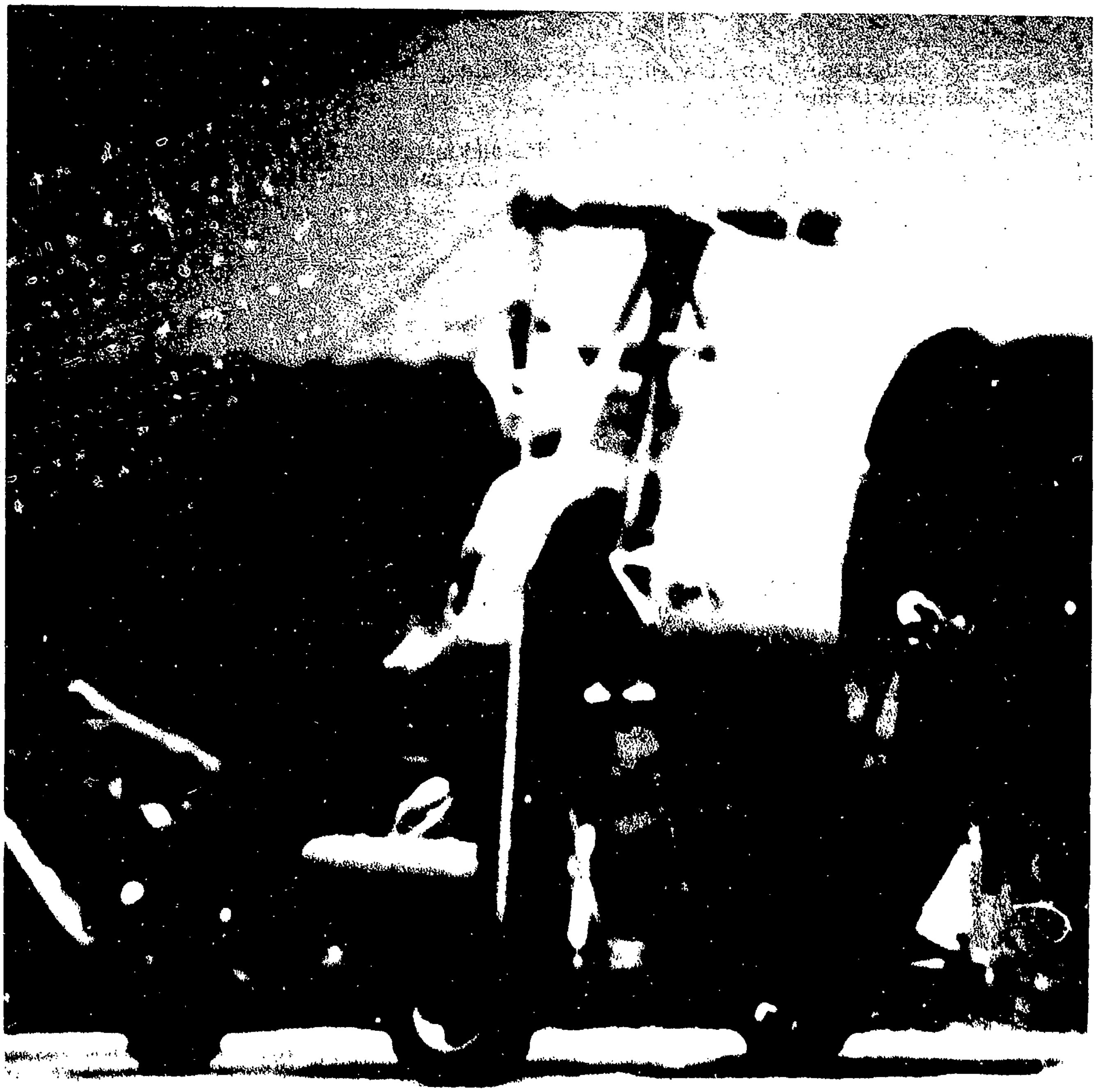

Figure B-1. Irish Flyer I

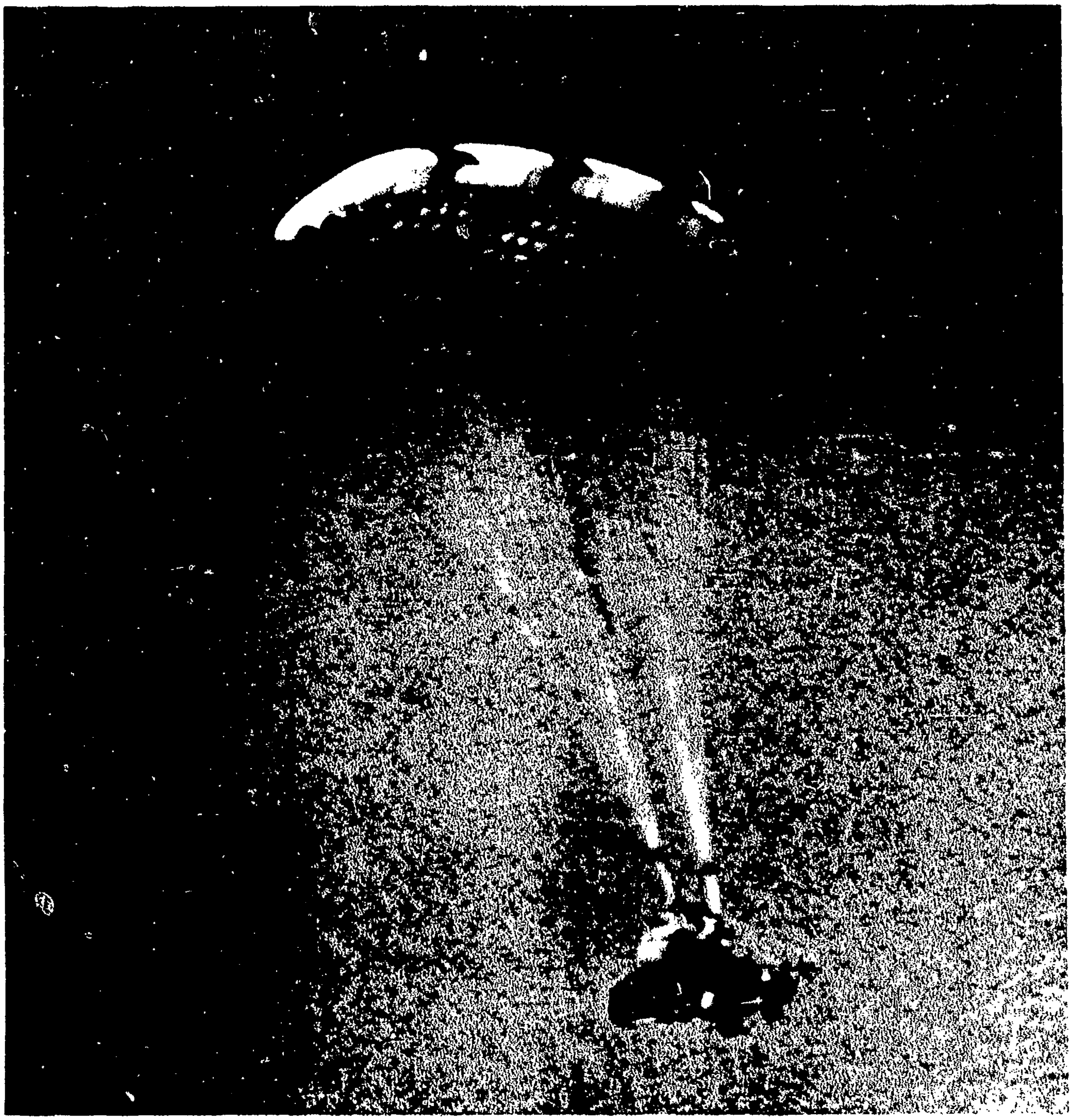

Figure B-2. Flight of Irish Flyer I

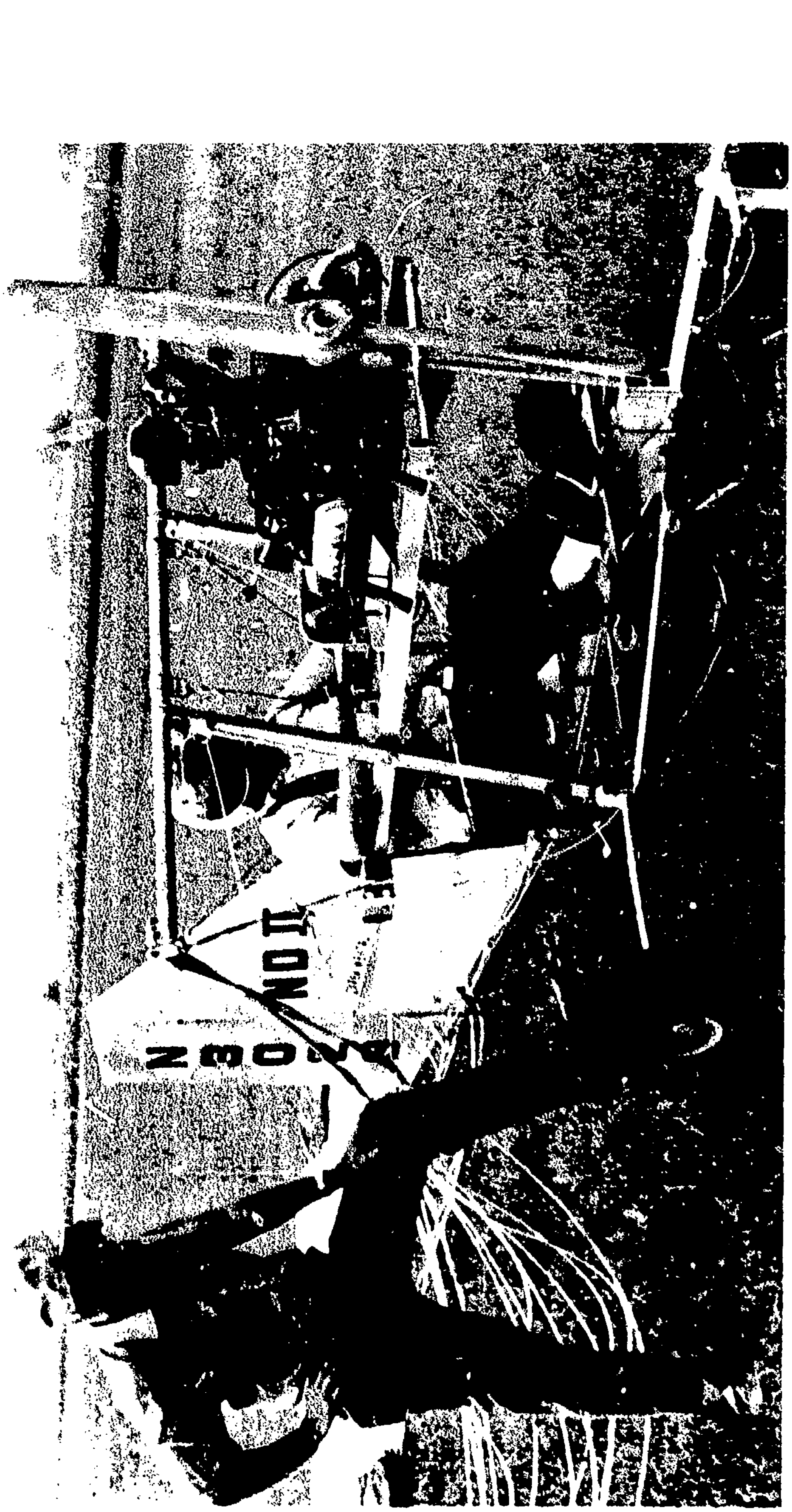

Figure B-3 Irish Flyer II

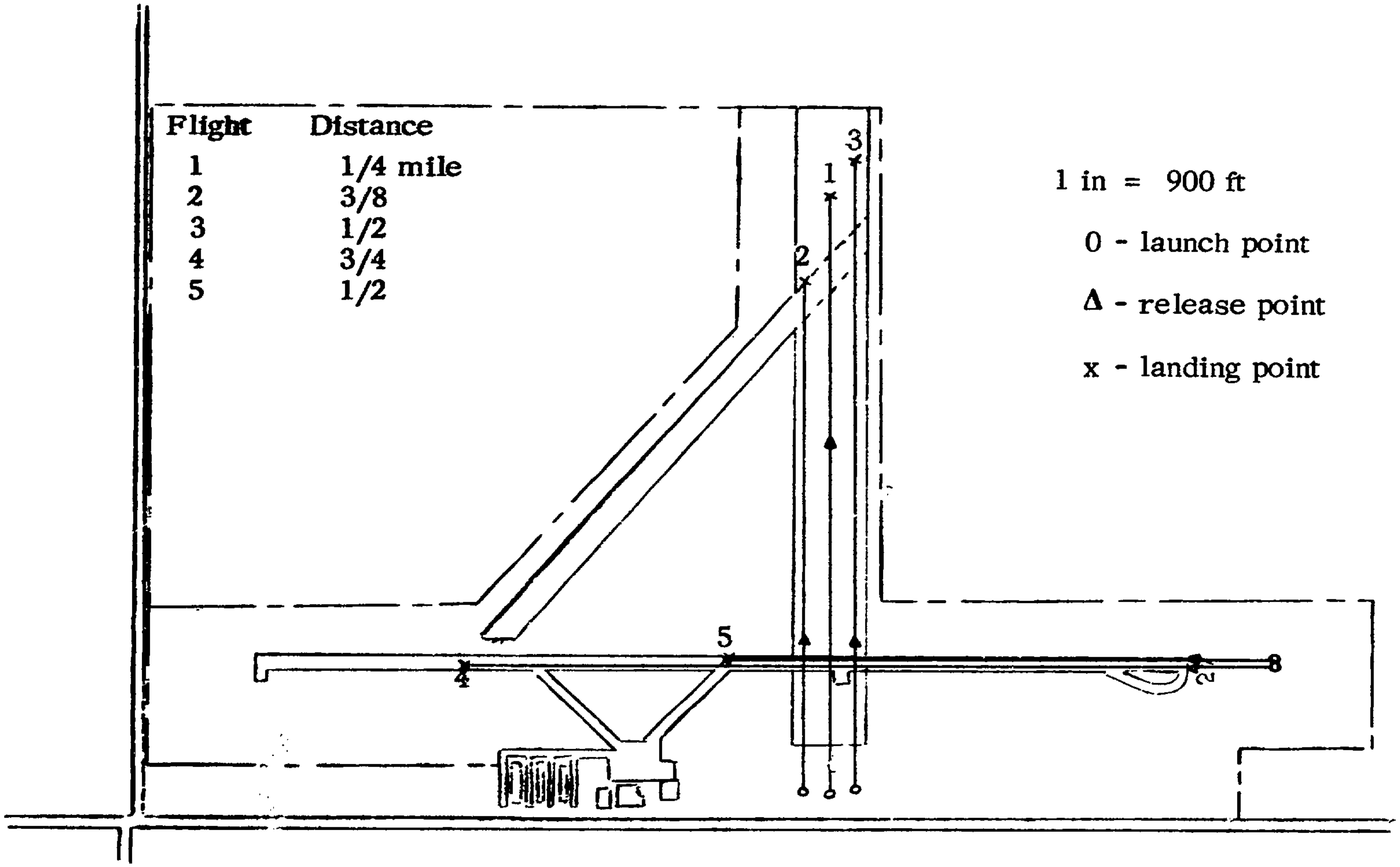

Figure B-4. Flights of Irish Flyer II.

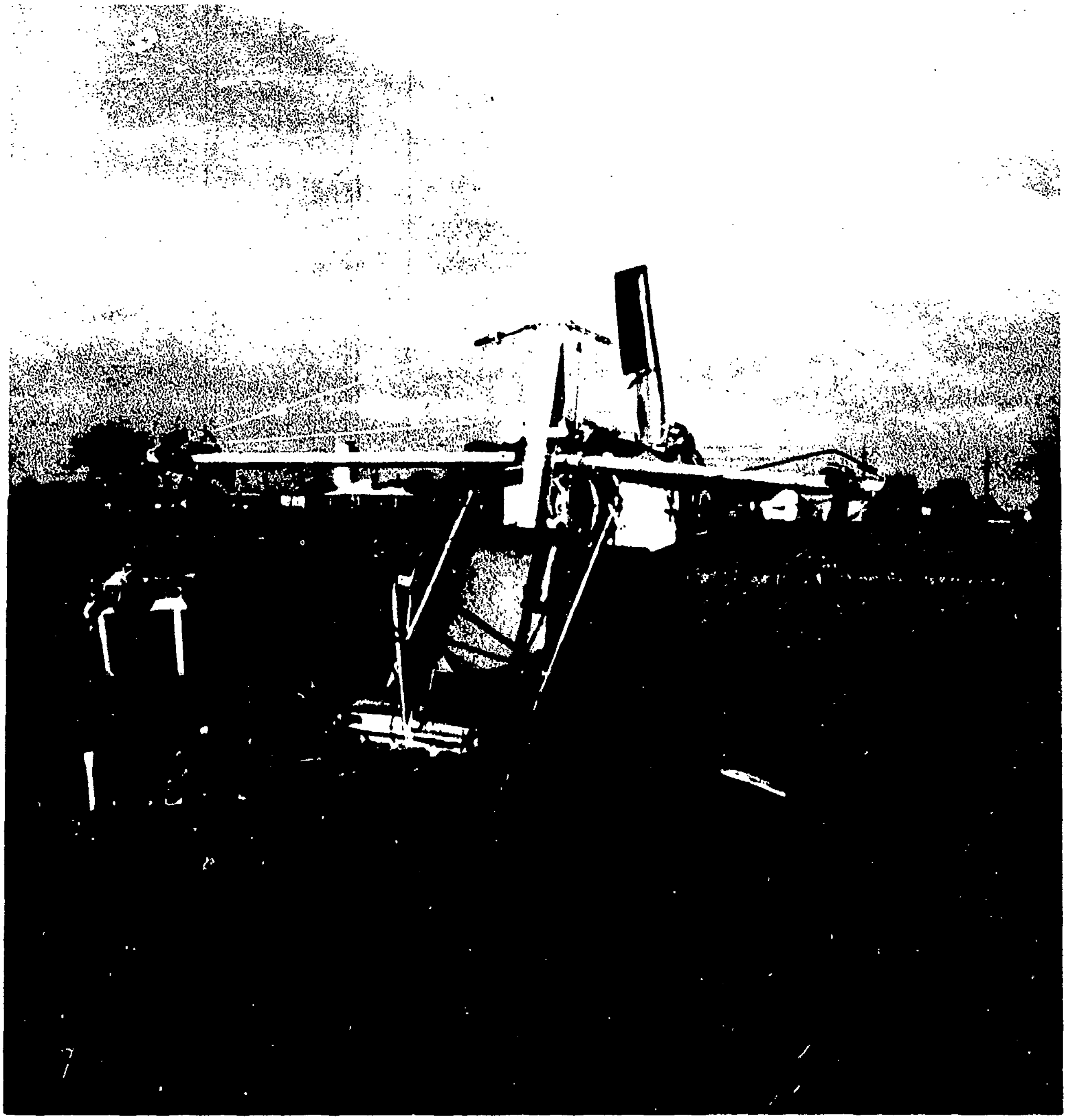

Figure B-5. Irish Flyer III.

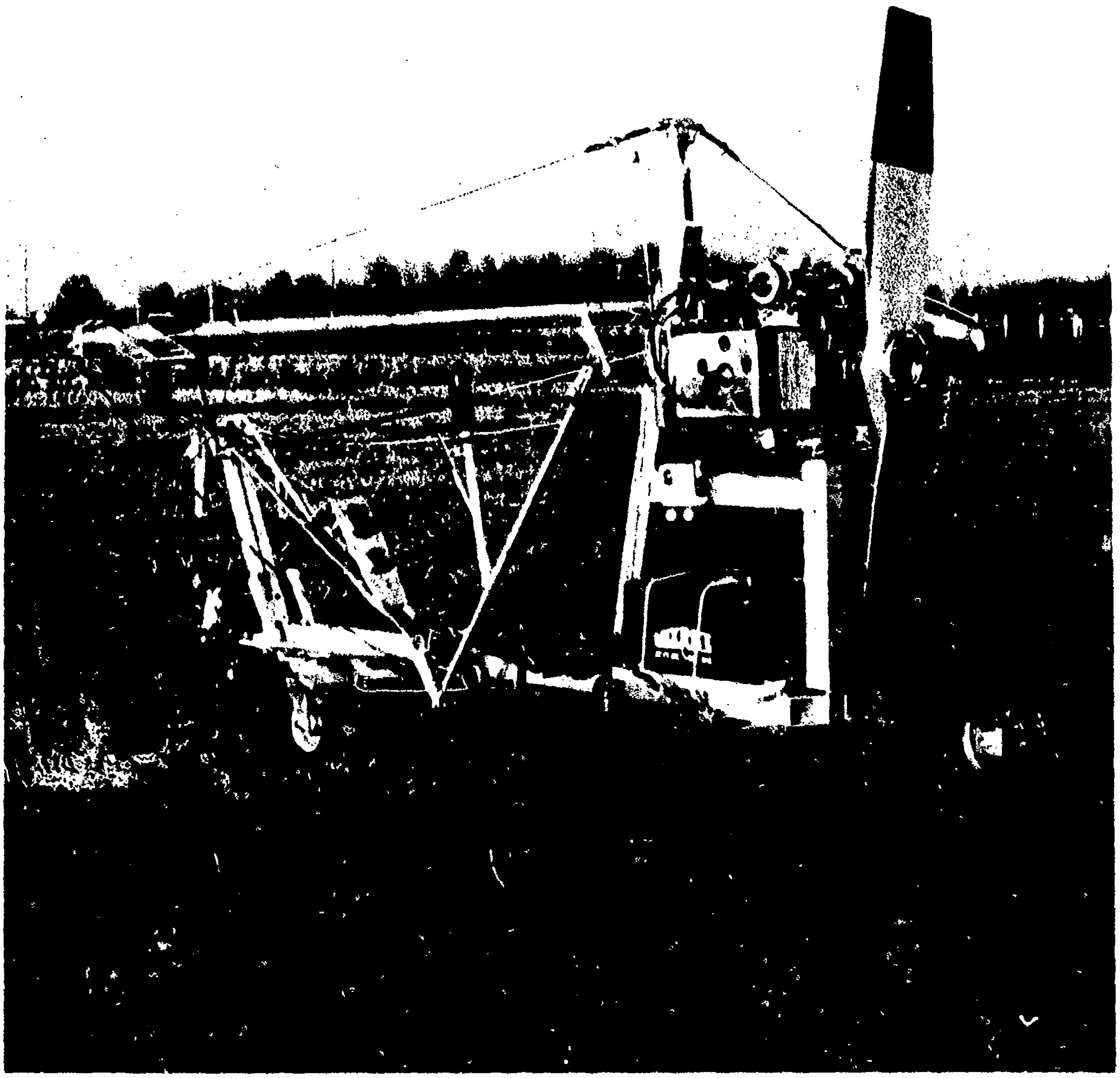

Figure B-6. Irish Flyer III

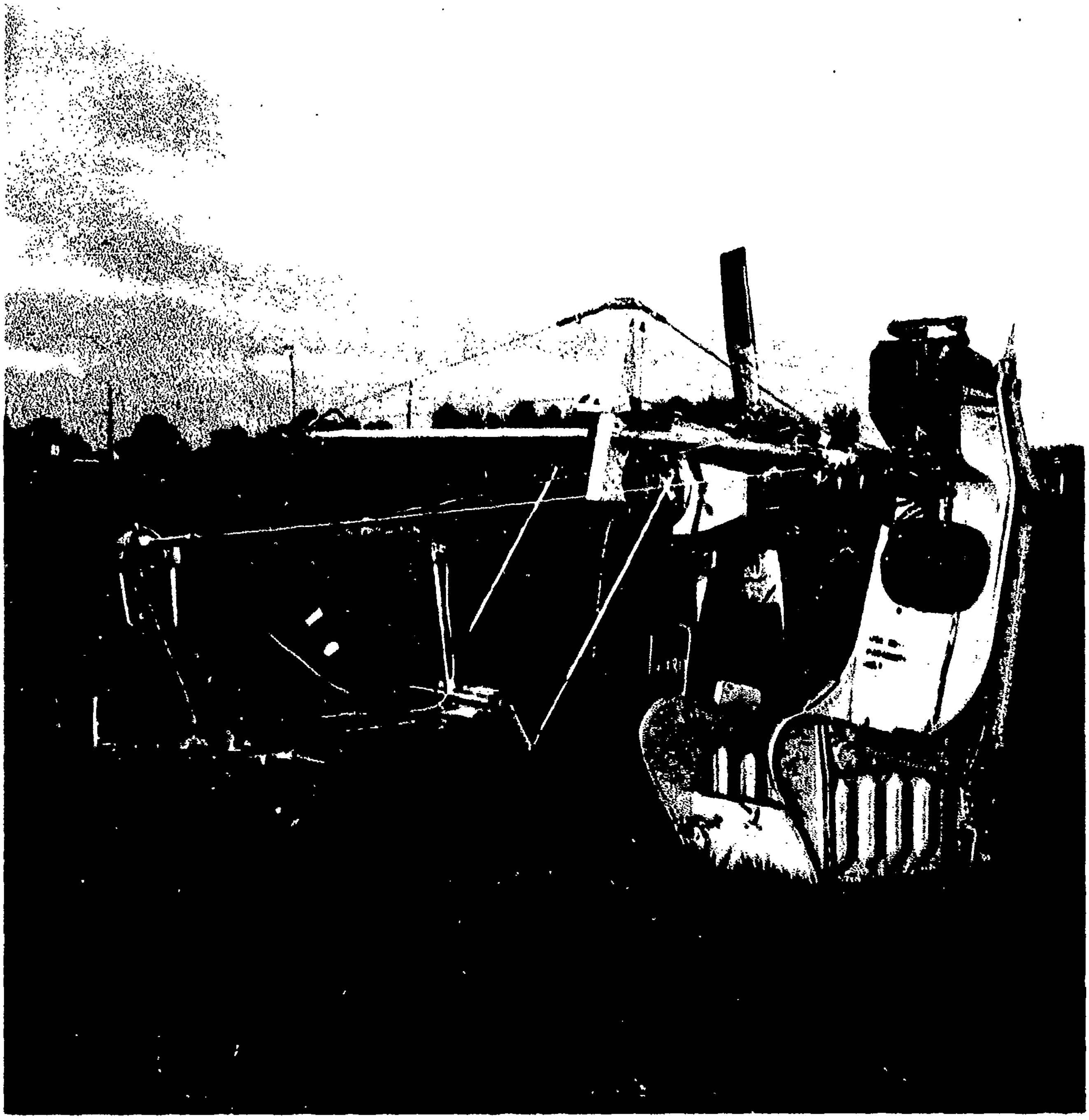

Figure B-7. Pilot Ejection Seat Flight Capability.

Figure B-8. Direct Tow Parafoil Stability and Control Ground Tests.

Figure B-9. Flight of Irish Flyer III.

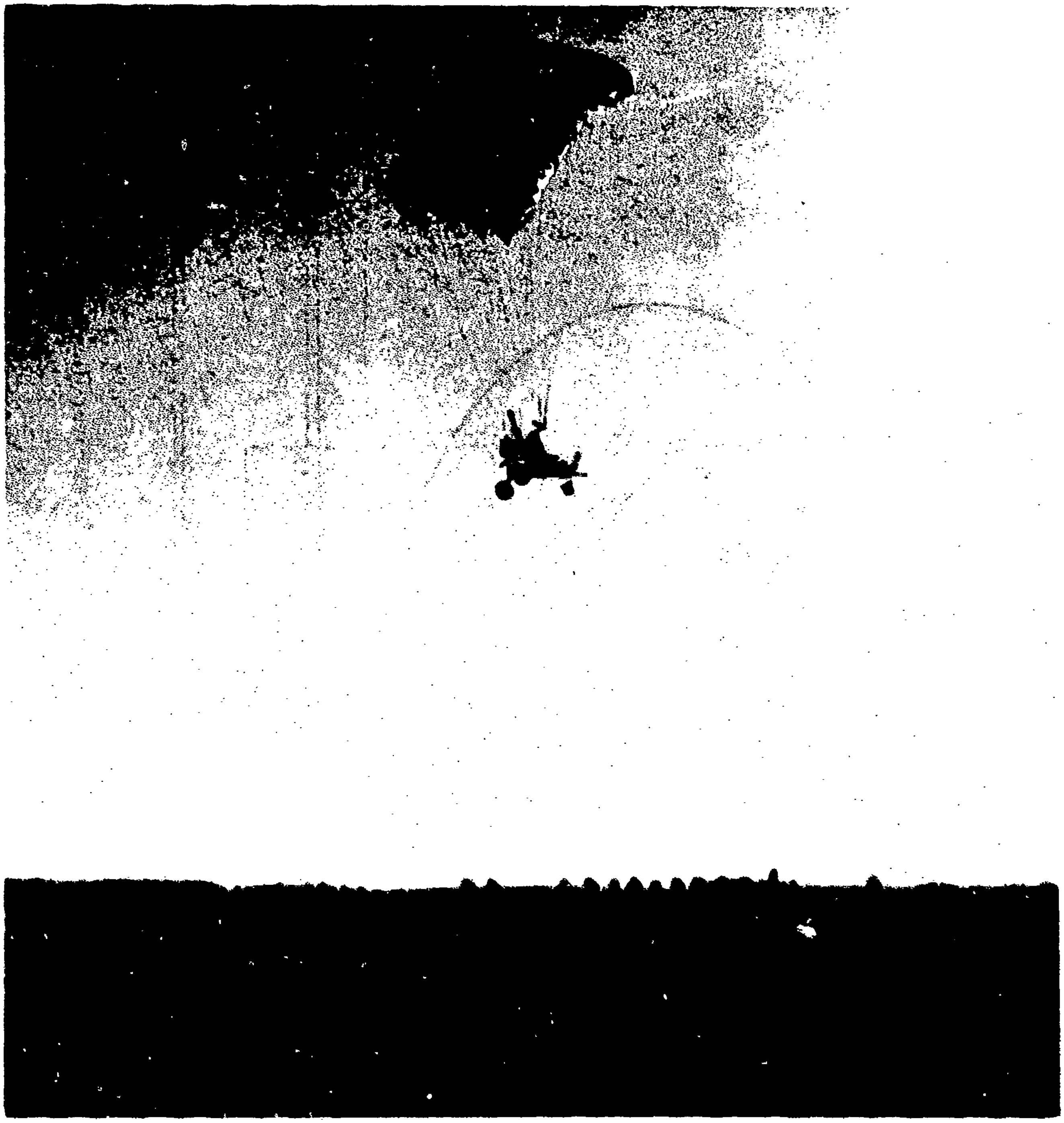

Figure B-10. Landing Approach on Flight of Irish Flyer III.

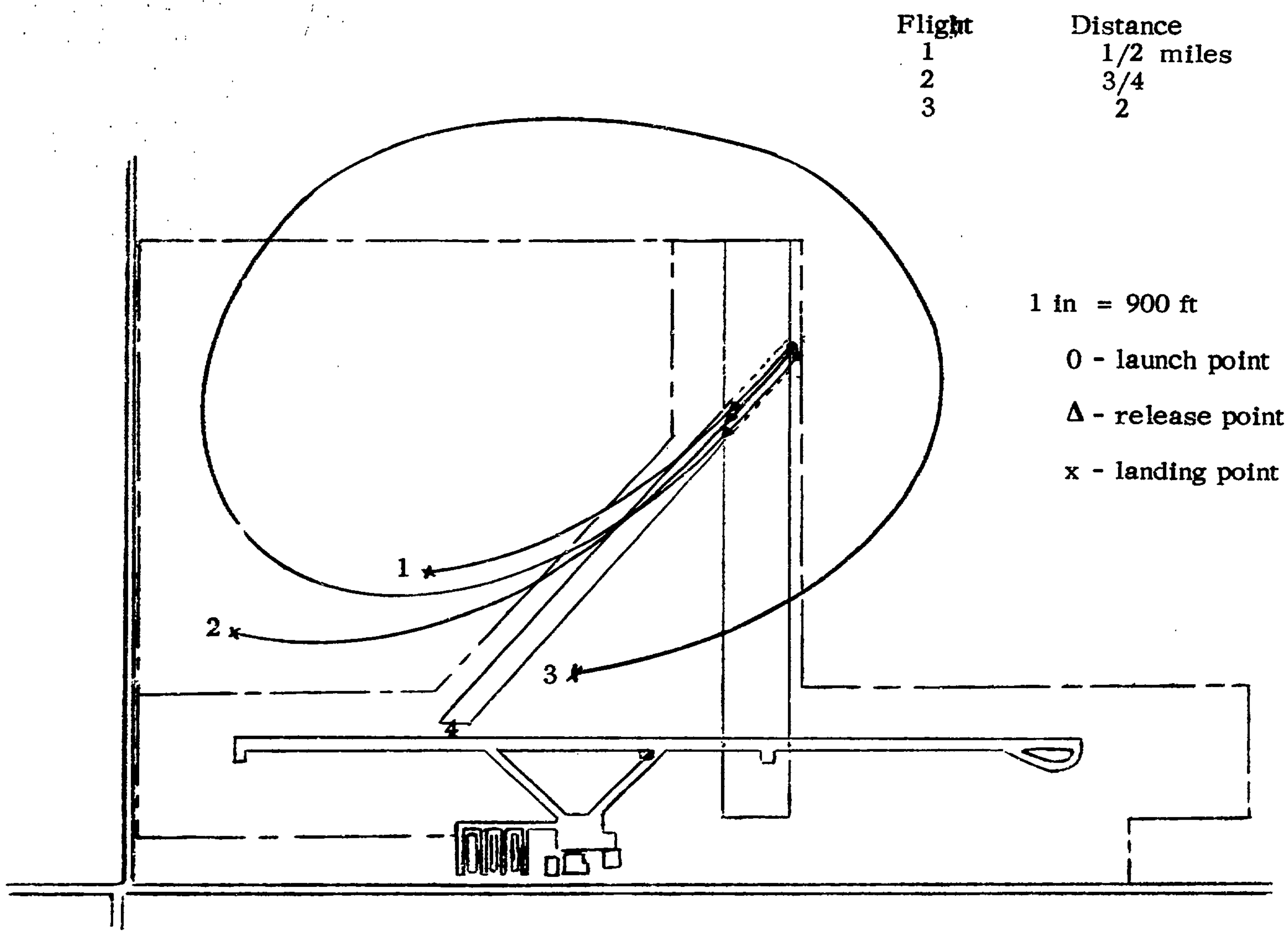

Figure B-11. Saturday Flight of Irish Flyer III.

Flight Distance = 12 miles

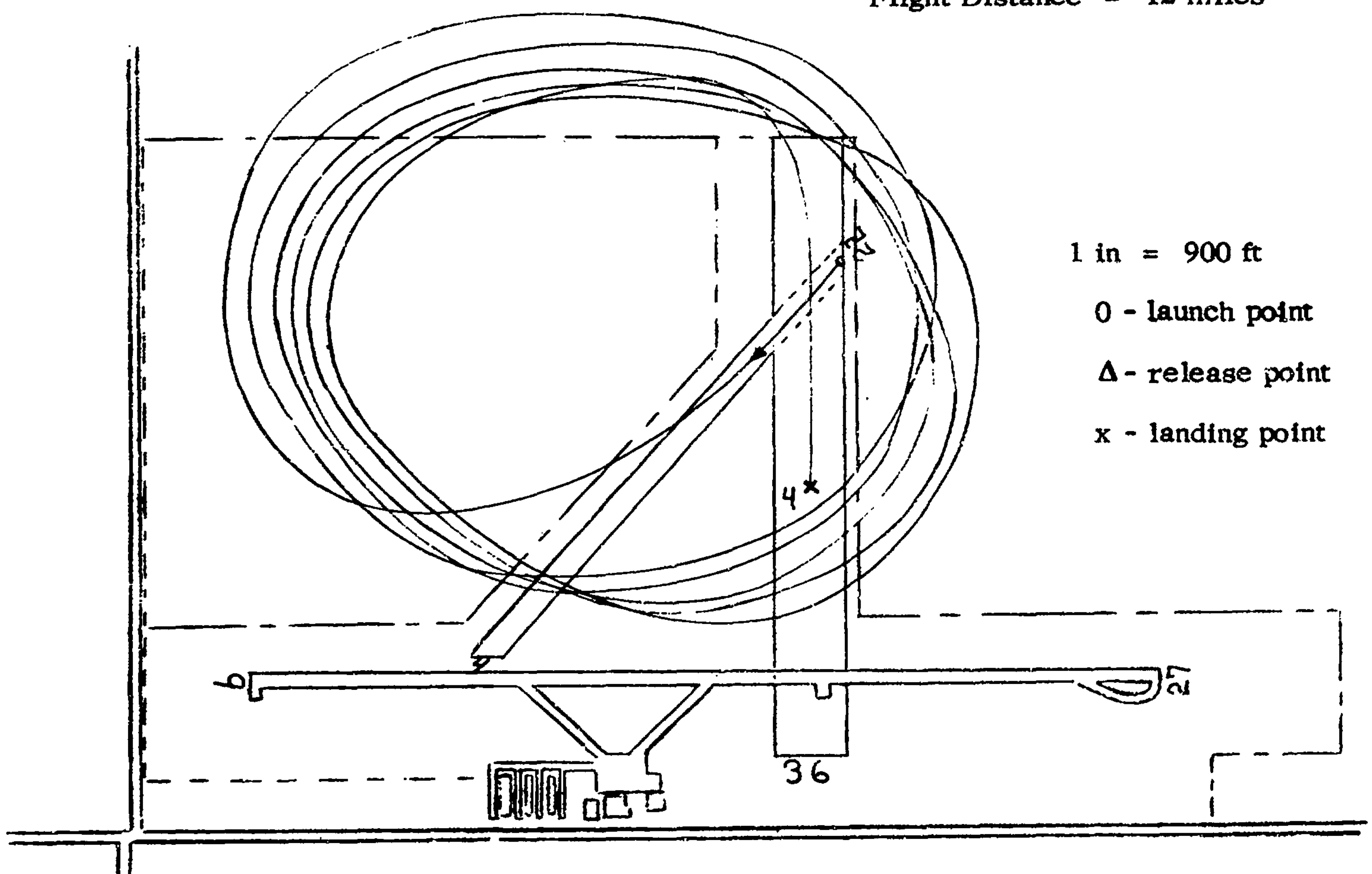

Figure B-12. Sunday Flight of Irish Flyer III.

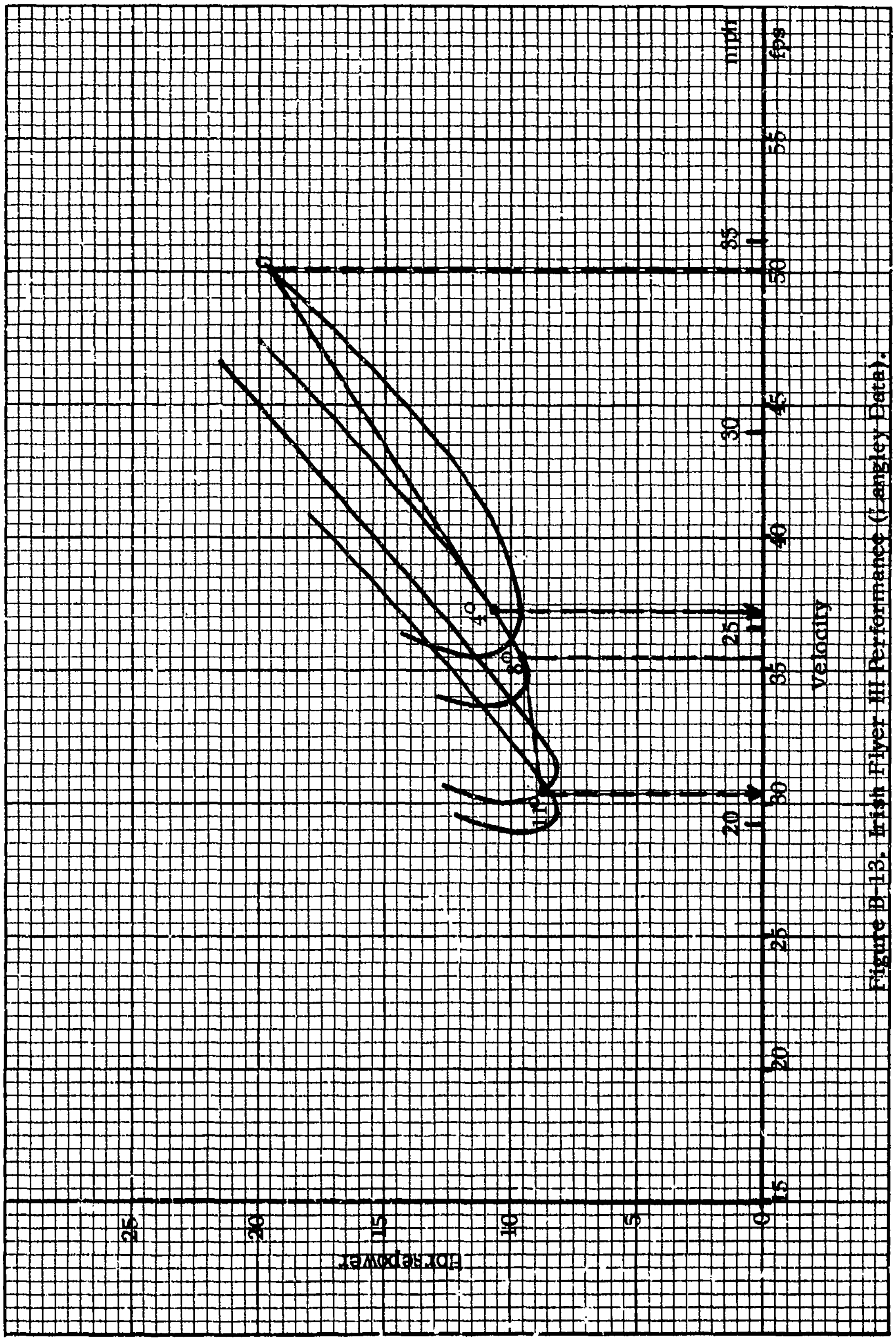

Figure B-13. Irish Flyer III Performance (Langley Data).

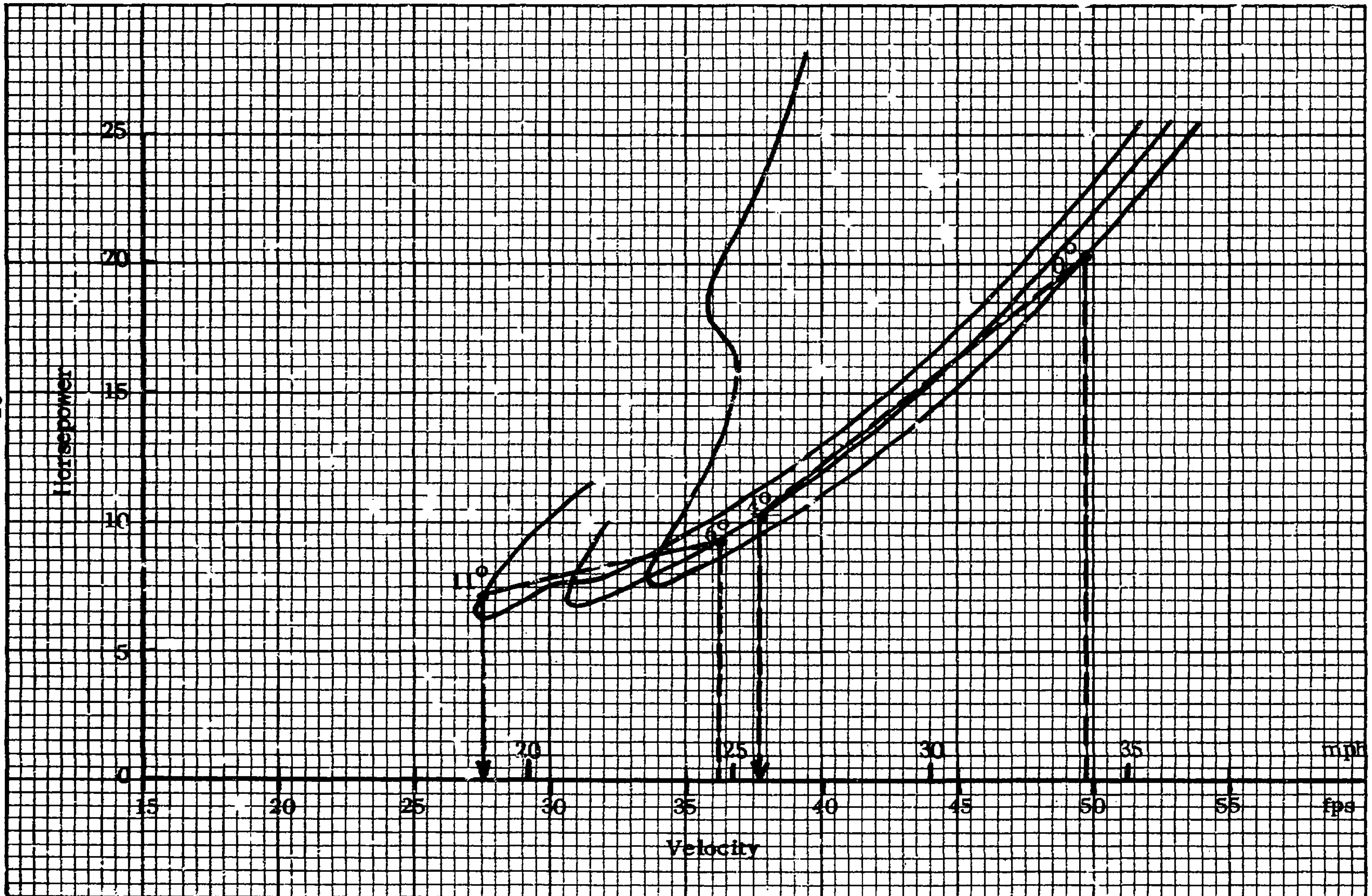

Figure B-14. Irish Flyer III Performance (Notre Dame Data).

α	C_L	C_D*	C_D**	
3	.694	.376	.326	2/3 Flaps $\Delta C_D = .076$
4	.800	.374	.324	
5	.856	.376	.326	
6	.898	.378	.328	
7	.927	.385	.335	
8	.951	.392	.342	
9	.969	.405	.355	
10	.987	.418	.368	
11	1.000	.438	.388	
3	.969	.438	.388	Full Flaps $\Delta C_D = .076$
4	1.000	.438	.388	
5	1.016	.440	.390	
6	1.036	.440	.394	
7	1.065	.453	.403	
8	1.069	.462	.412	
9	1.082	.480	.430	
10	1.093	.498	.448	
11	1.109	.525	.475	
1	.475	.252	.202	0 Flaps $\Delta C_D = .076$
2	.527	.252	.202	
3	.560	.256	.206	
4	.600	.260	.210	
5	.625	.272	.222	
6	.660	.283	.233	
7	.682	.294	.244	
8	.702	.307	.257	
9	.716	.320	.270	
10	.729	.336	.286	
11	.736	.356	.306	
12	.738	.372	.322	
13	.733	.388	.338	
14	.727	.407	.357	
15	.702	.432	.382	

Figure B-15. Aerodynamic Coefficients for Flap Deflection (NASA).

*The NASA data[14] was all increased by $\Delta C_D = .076$ to account for the added drag due to flight payload.

**The resulting total drag was then reduced by $\Delta C_D = .05$ to allow for the improved flight rigging and Parafoil configuration.[14]

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